Stress-Life and Crack Growth Calculations on an Aircraft Main Frame Section Using MSC.Fatigue

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Summary

The main objective of this project was to perform Stress-Life and crack growth calculations on a typical aircraft main frame section using MSC.Fatigue; Thereby demonstrating the possible usefulness of MSC.Fatigue as a general-purpose fatigue analysis tool. The project was based on a medium range passenger aircraft. This fatigue analysis software was used in conjunction with Nastran, Patran and Marc.

The following tasks were undertaken.

- Set up model and check model quality (mesh density etc)
- Perform Stress-Life fatigue analysis with MSC.Fatigue
- Identify critical location
- Investigate stress conditions in this region (MARC/Nastran)
- Demonstrate current state-of-the-art techniques for dealing with fracture, stress intensity function calculations and crack propagation analysis.
- Perform crack growth calculation with MSC.Fatigue
- Perform sensitivity studies related to load level

An Overview of FEA Based Fatigue Design

In fatigue design, three core fatigue methodologies have become established. The first two techniques that were developed do not model the crack growth process at all. Instead, they use the concept of similitude to determine the number of cycles to failure; 'Failure' being defined as some predetermined crack length, or loss of stiffness, or separation of the component being designed. This means that the relationship between component life and load level in a test specimen can be compared directly with that expected in service (assuming the component is tested under identical conditions). The first of these two methods is based on stress, the so-called ***Stress-Life*** (S-N, nominal stress, or total life) method. The second, and more recent technique is based on strain, the so-called ***Strain-Life*** (Local-Stress-Strain, Crack-Initiation, Manson-Coffin or Critical-Location Approach - CLA) method.

The third, and most recently developed method deals with ***Crack-Propagation*** and relies on the observation that once cracks become established they have a stable growth period. This is usually described using linear elastic fracture mechanics. It further relies on the assumption that crack growth rates are proportional to the applied stress intensity (a function of crack length, geometry and stress level).

An Overview of the FEA Based Fatigue Environment

MSC.Fatigue is the leading tool for FEA based fatigue design and Figure 1 gives an overview of the MSC.Fatigue FEA based fatigue environment. The three plots on the left indicate the FEA results, applied loading and materials information. The three plots on the right show the types of result visualisation that are possible. The centre box indicates the types of fatigue calculations that can be done. All of the fatigue techniques specified in Figure 1 are completely, or substantially, based on one of the three standard life estimation methods, i.e., Stress-Life, Strain-Life or Crack-Propagation, described in detail later.

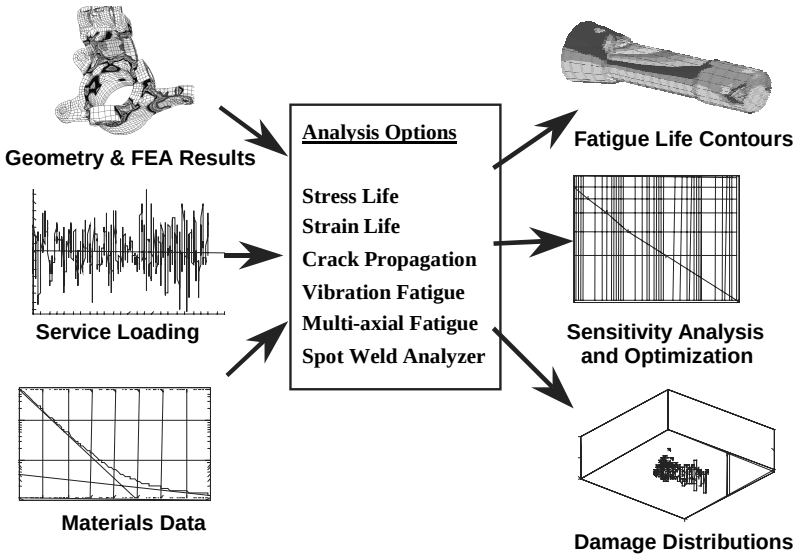


Figure 1. An overview of the MSC.Fatigue FEA based fatigue environment

In aircraft applications, subjected to mainly High Cycle Fatigue (HCF) situations the data widely available is of the S-N form. This study therefore concentrates on the Stress-Life, rather than Strain-Life approach in conjunction with a crack growth analysis. The modules of MSC.Fatigue used in this study are therefore "Basic" (S-N), "Fracture" (crack growth analysis) and "Utilities" (for all manipulation of data)

This report contains only a limited amount of theory relating to fracture, fatigue and crack propagation. Any readers wishing to obtain more detailed information are referred to any one of the standard textbooks on the topic.

Set Up Model

The FE model used to represent a typical main frame section is shown in **Figure 2**. This is a small part from the complete main frame positioned approximately in line with the main wheels. The model included all of the boundary conditions and Nastran base load case (stress) results. Altogether 32 load cases were required in order to define the complete mission profile and Table 1 lists these base load cases. Most of the loads have 2 elements that, together, create a fully reversed loading sequence. **Figure 3** to **Figure 7** show some typical base load case stress results obtained with Nastran.

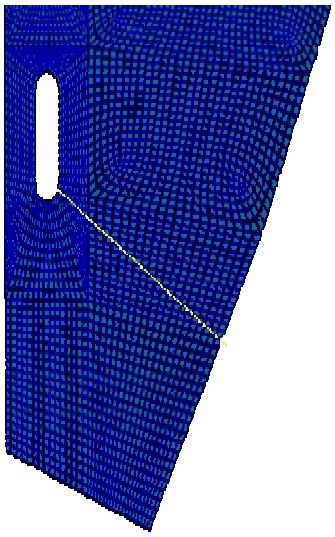


Figure 2. FE model (showing critical plane?)

Table 1. Unit Stress Loading Cases to be Used in Fatigue Analyses

Load Condition	***Cycles per flight***	***Fully reversed***
Unloaded	1/2	Transition
Ground Turn	6	Fully reversed
Taxi	7	Fully reversed
Rotation	1/2	Transition
Flaps Down Departure +/- Vertical Gust	2	Fully reversed
Flaps Down Departure +/- Lateral Gust	2	Fully reversed
Final Climb +/- Vertical Gust	4	Fully reversed
Final Climb +/- Lateral Gust	4	Fully reversed
Cruise +/- Vertical Gust	4	Fully reversed
Cruise +/- Lateral Gust	4	Fully reversed
Initial Descent +/- Vertical Gust	4	Fully reversed
Initial Descent +/- Lateral Gust	4	Fully reversed
Flaps Down Approach +/- Vertical Gust	3	Fully reversed
Flaps Down Approach +/- Lateral Gust	3	Fully reversed
Yaw Manoeuvre	12	Fully reversed
Flare	1/2	Transition
Touch Down	1	Fully reversed
Unloaded	1/2	Transition

The objective of any mission profile definition is to ensure that these loads are applied with the correct magnitude, and in the correct sequence. In order to get the results in to SI (MPa) units it was necessary to multiply all stress results by 10. This was done in the mission profile definition (see later). One very important issue was the nature of the stress conditions. For instance, was the crack being driven by tensile or shear cracking? This issue is also addressed in more detail later.

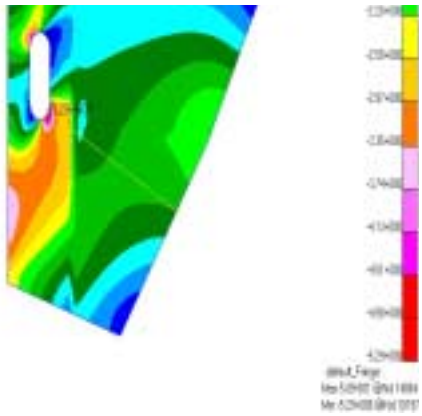


Figure 3. Static stress response GT-15

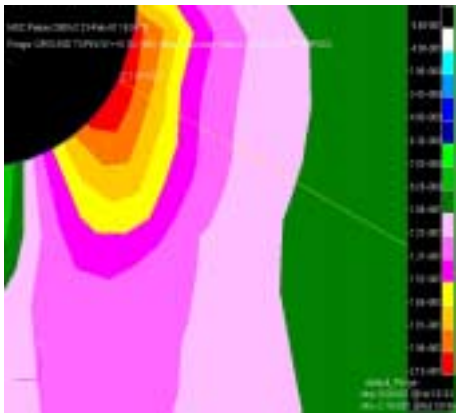


Figure 4. Static stress response GT+15

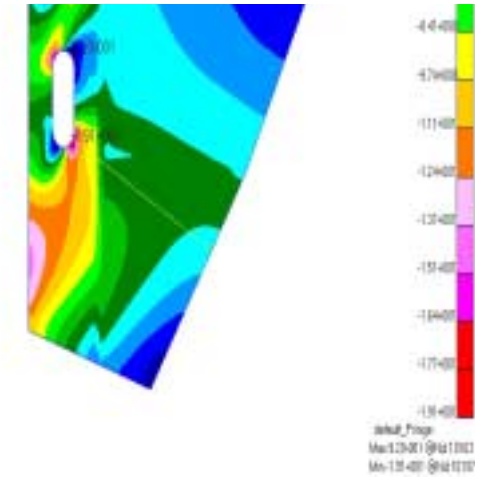


Figure 5. Static stress response Taxi 1.3

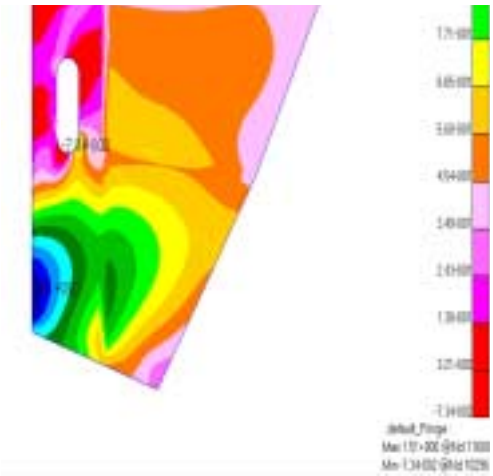


Figure 6. Static stress response Flap Down Dep

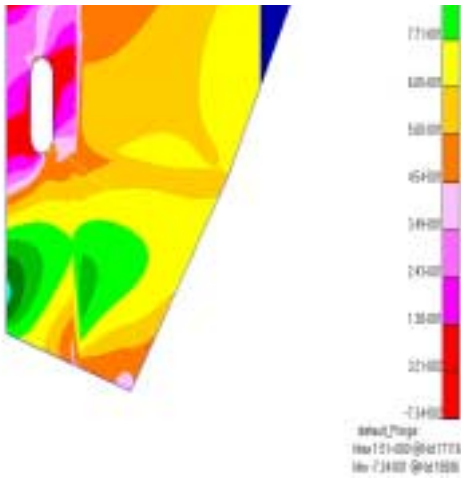


Figure 7. Static stress response Cruise Lateral Gust

Mesh Refinement

In order to check model quality in terms of mesh refinement a small “cut out” from the critical region in the basic mesh model provided was refined to the order of 4x, 8x and 16x. The stress results in the critical locations were observed to converge to a value 25% higher than from the standard mesh. For all subsequent analysis the standard mesh was used and the stresses obtained were multiplied by 25% (ie by 1.25 during the mission profile definition).

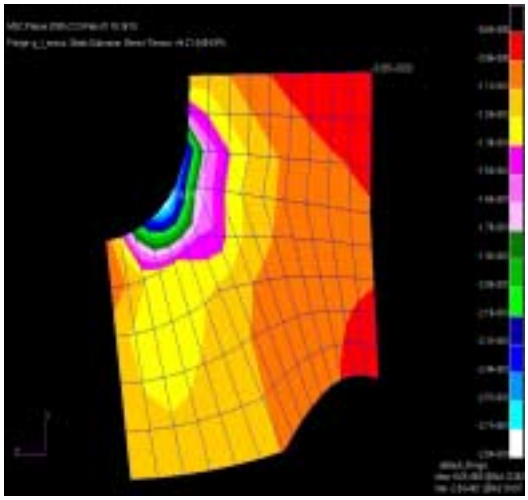


Figure 8. Mesh x 1

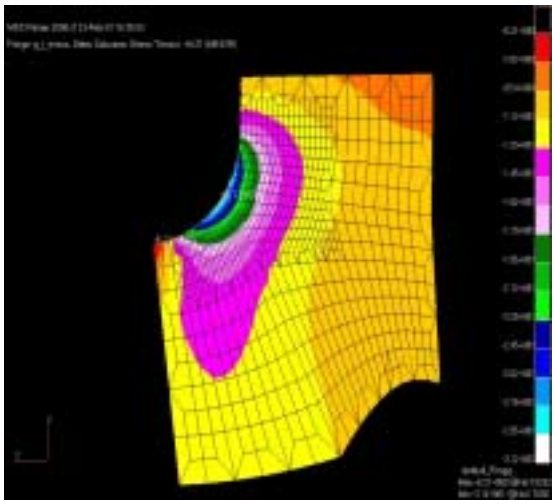


Figure 9. Mesh x 4

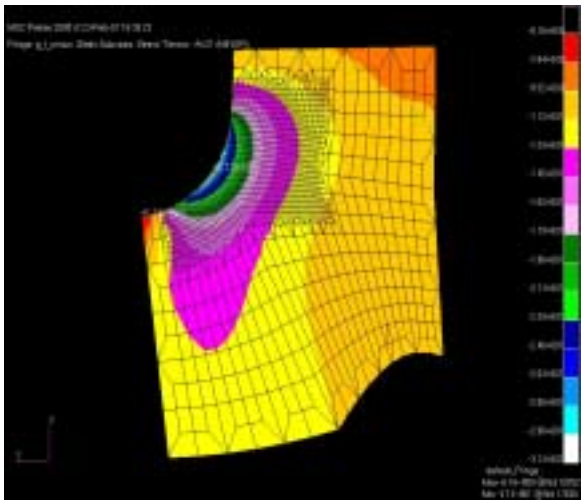


Figure 10. Mesh x 8

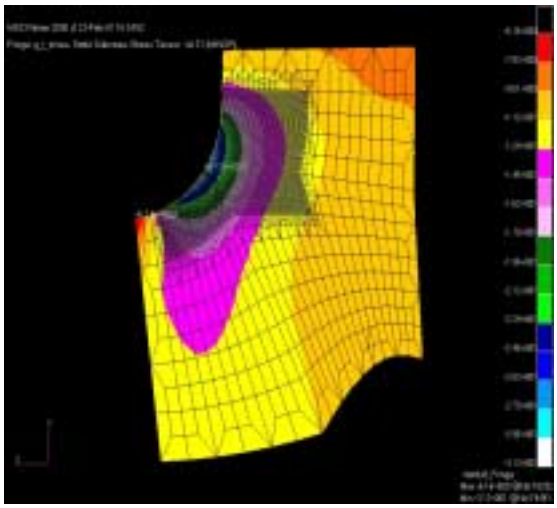


Figure 11. Mesh x 16

Definition of Mission Profiles in MSC.Fatigue

It is not obvious, with MSC.Fatigue, how to set up and apply multiple loadings of the form normally applied to aircraft structures. Table 1 lists the various base load (stress) cases which must be applied. These loadings should be applied sequentially, one at a time. However, MSC.Fatigue expects multiple loadings to be applied simultaneously. The linear superposition of these loadings is then usually achieved with the following equation,

$$\sigma_{ij}(t) = \sum_{k} P_k(t) \left(\frac{\sigma_{ij,k}}{P_{k,fea}} \right)$$

where $P_k(t)$ is the force time history, $P_{k,fea}$ is the magnitude of the force used to produce the static load case (usually unity) and $\sigma_{ij,k}$ is the Nastran static stress result at point ij, for load case k. For the purpose of the work reported in this study the mission profile was easily defined by setting up, for each load case, a time series of zeros and ones, with the base load case being applied by specifying 1, and being turned off by specifying 0. These time series $P_1(t)$ to $P_{32}(t)$ are shown in the following figures. The resultant stress history at critical node

10186 is shown in **Figure 16**. It is important to note that there is always 1, and only 1, load case applied at any given time step.

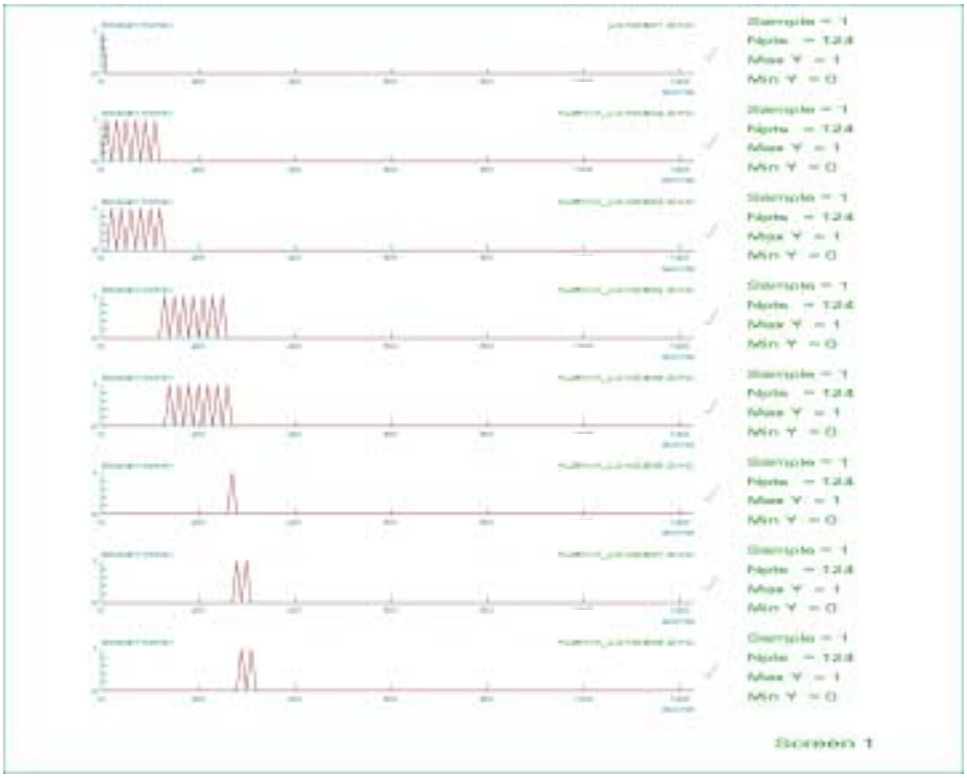


Figure 12. Applying load cases in MSC.Fatigue. Load cases 1-8

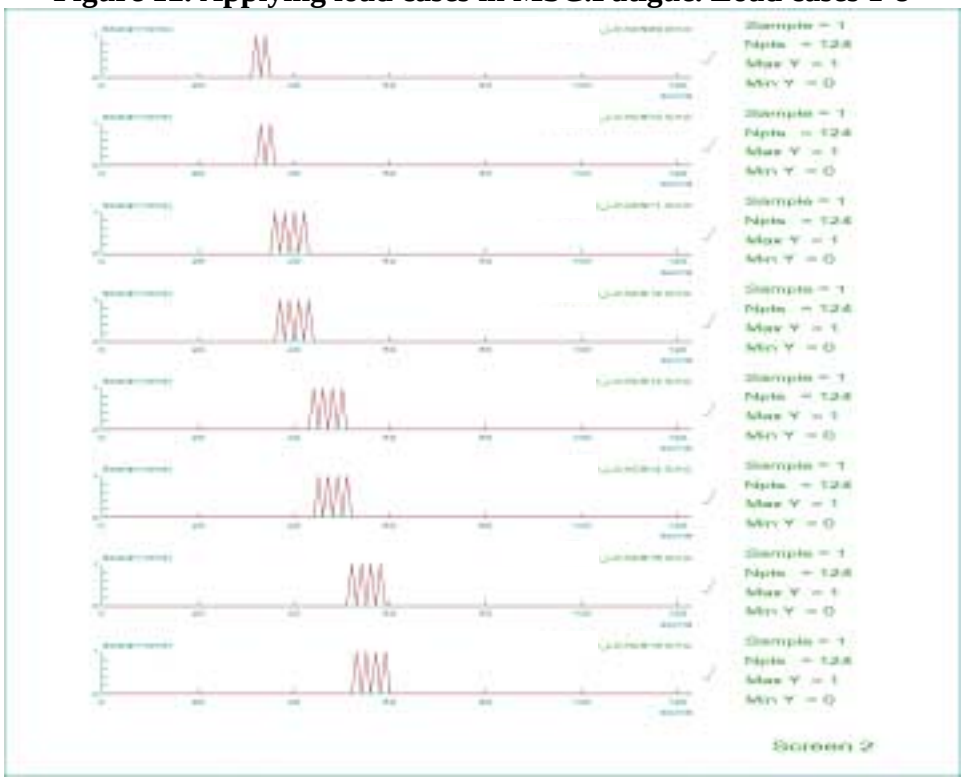


Figure 13. Applying load cases in MSC.Fatigue. Load cases 9-16

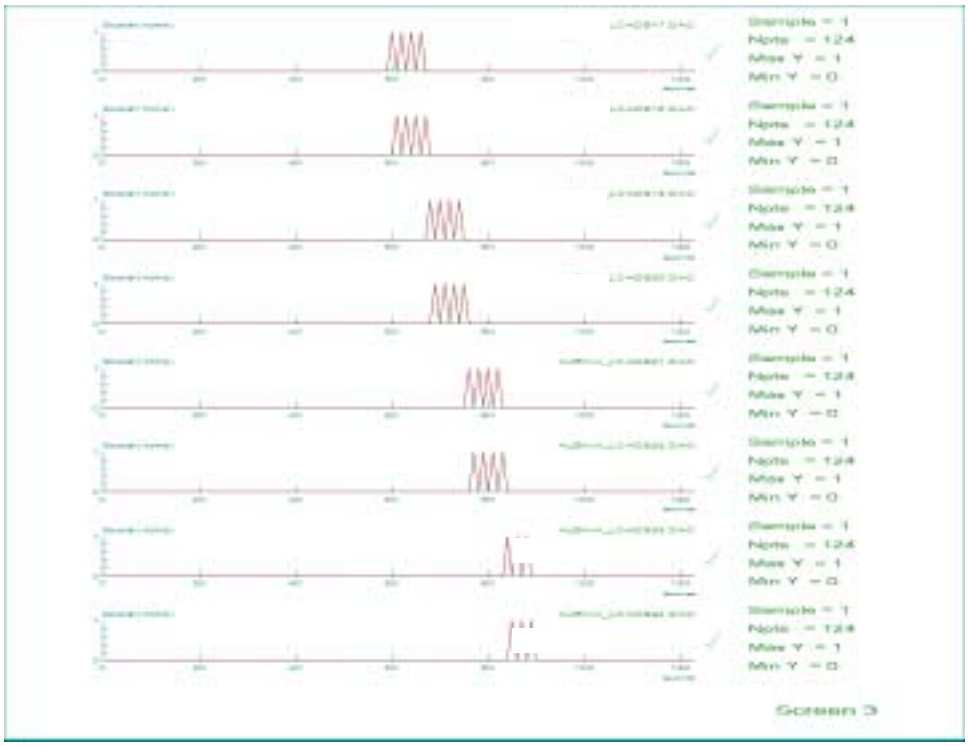


Figure 14. Applying load cases in MSC.Fatigue. Load cases 17-24

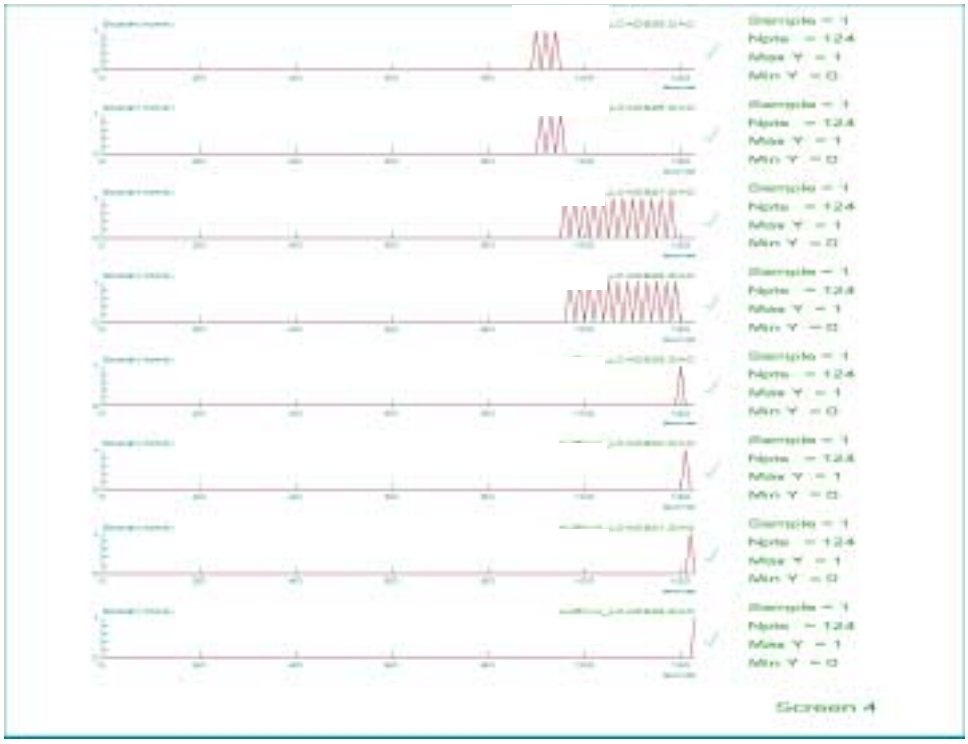


Figure 15. Applying load cases in MSC.Fatigue Load. cases 25-32

Stress-Life Fatigue Analysis

The basic set up procedure for performing a stress based fatigue calculation is given below. Most of these calculations can be done from within Patran, or "stand alone" from the command prompt. Most descriptions below assume the calculation is being done from within Patran unless otherwise stated. If a job has already been set up then a saved file (.FIN) will probably have been created in which all the job set up parameters will have been recorded and this can be read back in using ***Job Control → Read Saved Job.***

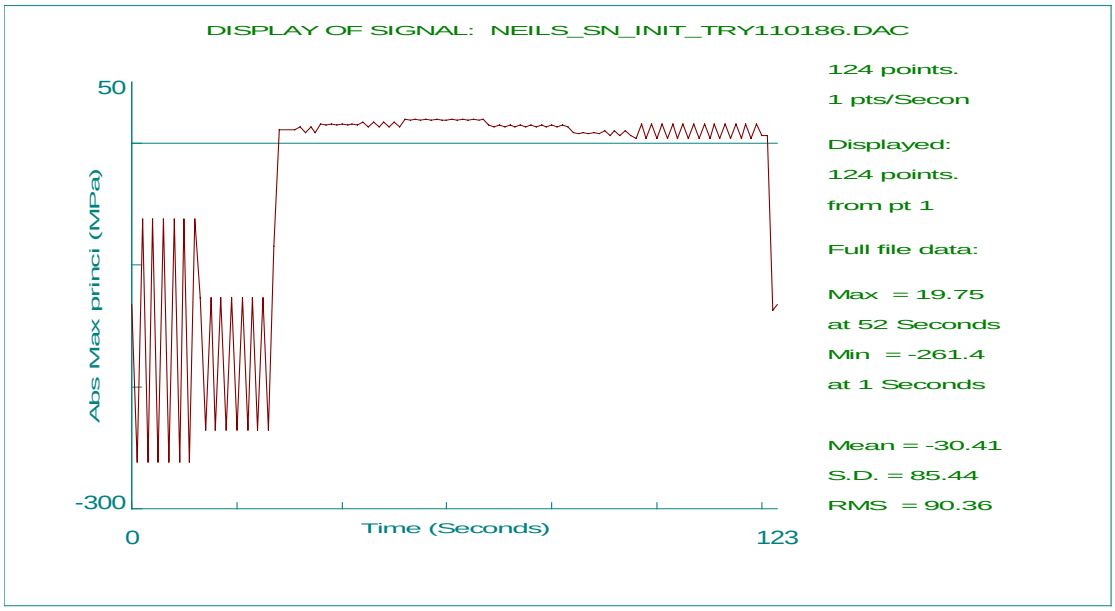


Figure 16. Principal stress obtained from critical location as a result of applying the 32 load cases simultaneously

The starting point is to run Patran, read in the appropriate model file and then spawn MSC.Fatigue using the ***tools*** top bar menu. This will spawn the form shown in Figure 17. This is the basic analysis form and should be worked through from top to bottom. Firstly, an S-N analysis type must be set, nodal (rather than element) averaging specified, global (rather than group) averaging chosen, and finally MPa chosen. Once these basic factors have been determined the 3 basic forms can be filled in.

Figure 17. The basic MSC.Fatigue form

Firstly the ***Solution Parameters*** form needs to be filled out as in **Figure 18**. In this form it must be decided if any mean stress correction is to be applied. Since all mean stresses in this analysis are compressive it was decided to take a conservative approach and assume no beneficial effect, i.e. no mean stress correction was performed. The Max Abs Principal was used to compute stress results. This option picks the biggest principal stress whether it is compressive or tensile, thus ensuring ranges of stress are properly defined. A 50% certainty of survival (on material properties) was also defined in this form.

Figure 18. The Solution Parameters form

The second of the 3 main forms is the ***Materials*** form (see Figure 19). In this form the location of the materials database is defined and the various material properties applied to each region are defined. For this S-N part of the project the region "fatiguecalc" included the whole of the plate model shown in **Figure 2** (minus the MPC's). Also in this form any surface finish or surface treatments can be defined (not available for aluminium). Two sources of S-N material data were available. Firstly, an S-N curve (source no1) in the following form (see Appendix 1) was used.

$$N = 10^5 \left(\frac{600 - S\max(1+R)}{S\max(1-R)} \quad \frac{40.2}{200 - 40.2} \right)^{3.6}$$

"source no1" was the material Databank of Alenia Aerospazio Divisione Aeronautica, Pomigliano D'Arco Napoli.

This was produced for a Kt value of 3.1 and with R=0.06. The S-N curve, for these conditions, is shown in **Figure 20.** This plot shows both the above S-N curve equation and the single linear line fitted through it in MSC.Fatigue. A bilinear line could also have been used.

An S-N curve for the same material for Kt=1.0 was obtained from MSC.MVision for an R value of 0. This is shown in **Figure 21**. In order to make a comparison, the Source no1 supplied curve was normalised to a Kt=1.0 value and a comparison between these 2 sets of data, for Kt=1.0, is shown in **Figure 22**.

The last of the three main forms is the ***Loading*** form shown in Figure 23 and Figure 24. Here the "time histories" used to define the mission profile definition are associated with the correct Nastran base stress result and other scale factors and offsets are defined, if required. Excel was used to create asci load histories which were then read in to MSC.Fatigue using the asci to binary conversion utility in the tools+ pull down menu This requires a tools_pulldown.plb file to be installed on the system (see MSC.Fatigue installation notes).

This completes the solution set up procedure. The final task is to run ***Full Analysis*** from the ***Job Control*** button. Once this is finished the results then have to be read back in using the ***Results*** button at the bottom of the MSC.Fatigue form.

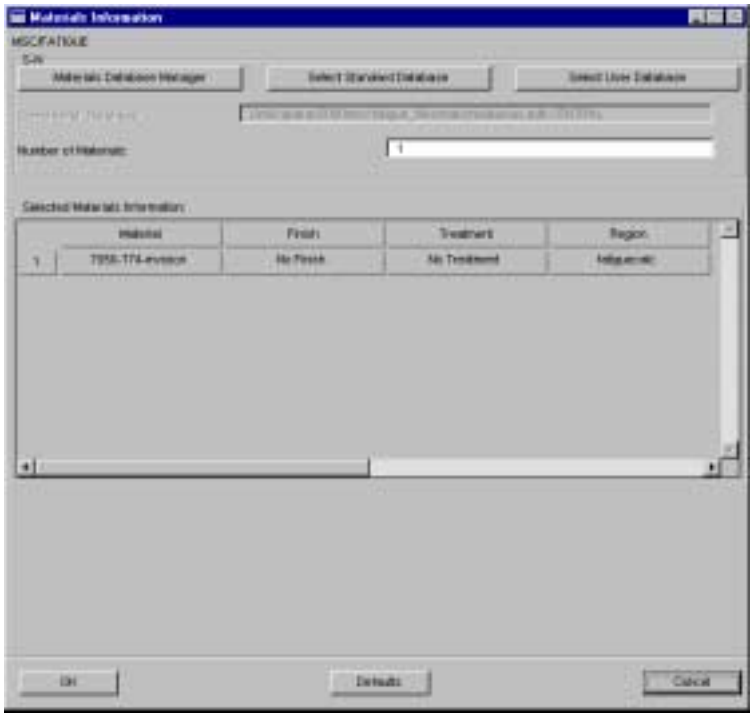


Figure 19. The Materials form

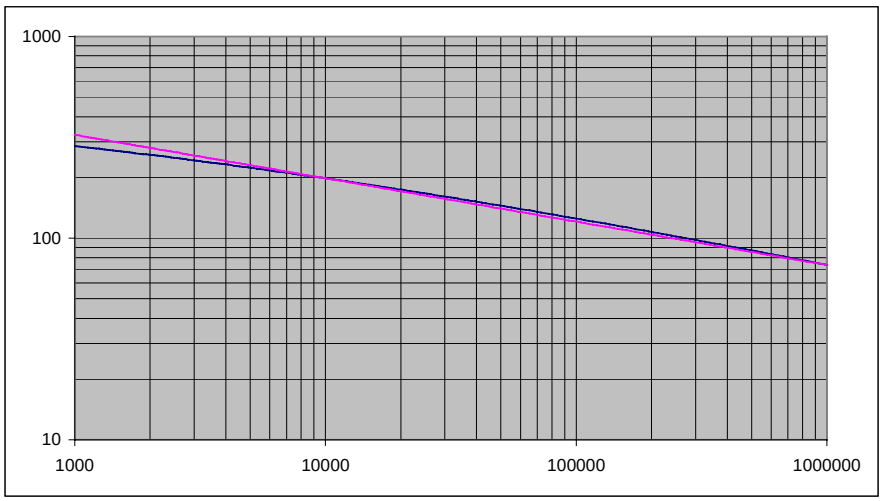


Figure 20. A comparison between the equation form of the (source no 1) S-N curve (7050 T74, shown in blue) and that fitted with a single linear line in MSC.Fatigue (shown in pink).

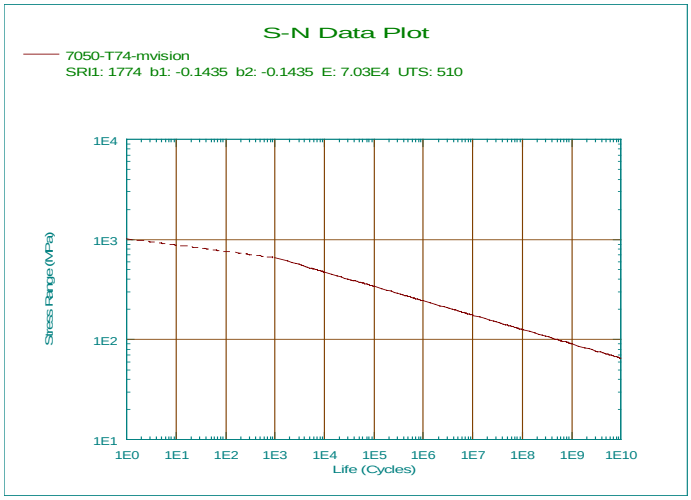


Figure 21. S-N data 7050 T74 obtained from MVision

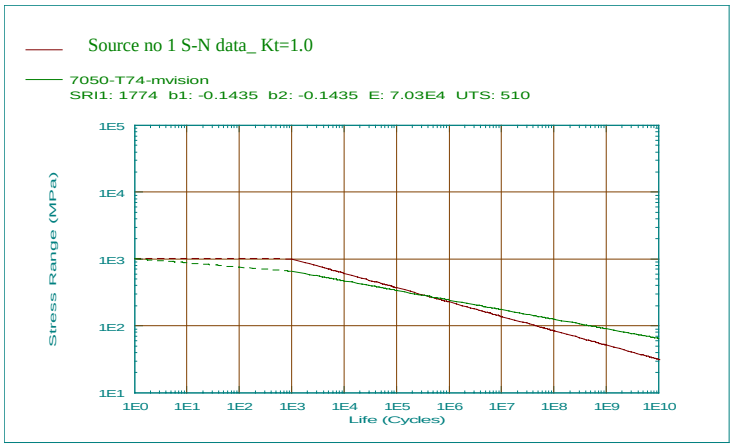


Figure 22. A comparison of the Source no1 supplied curve (normalised to Kt=1.0) with the MVision curve

Figure 23. The Loading form showing the cross referencing of Nastran load case (load case ID) and time history $P_k(t)$

Figure 24. The Loading form showing load magnitude (must be set to 1), scale factor and mean offset.

Identify Critical Location

Once the ***Full Analysis*** option was processed the results were read back in to Patran with the ***Read Results*** button. The result for damage using the MVision 7050 T74 S-N data is plotted in **Figure 25**, **Figure 26** and **Figure 27**. The critical location corresponds with node 10186. The fatigue life was recalculated at the critical location using the alternative Source no1 7050 T74 (Kt=1.0) S-N Curve. Both these results use the peak stress at node 10186 to calculate fatigue damage and hence life.

By choosing **Job Control → Interactive** the form shown in Figure 28 is spawned which provides a number of useful post processing options, including a ***Results Processing*** option (see Figure 29) which reads in the .FEF results file. Using this option the table of most damaged nodes (**Figure 30**) and single shot result for node 10186 (Figure 31) are shown for the MVision 7050 T74 data. Note that there is a slight difference between the result in Figure 31 and Figure 32 caused by a very small change in the S-N material specification used.

As a comparison, a "reference" stress was calculated near the critical location (see **Figure 59**, and Figure 60). The location for this region was determined after careful scrutiny of the stress distributions shown in Appendix 4. The fatigue calculation was then repeated using this reference stress and the Source no1 7050 T74 data with Kt=3.1. The result is shown in Figure 32.

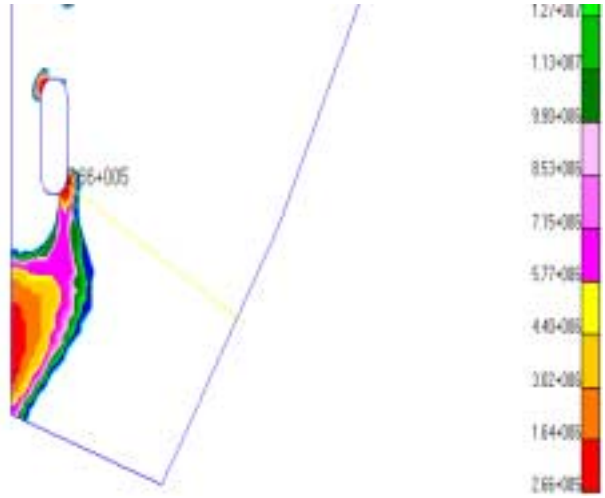


Figure 25. Plot of total life

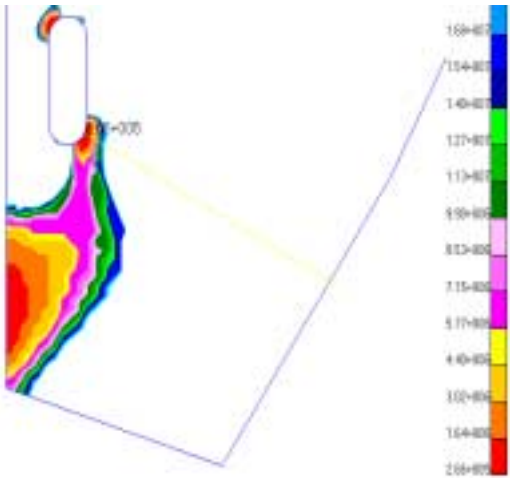


Figure 26. Plot of total life zoomed

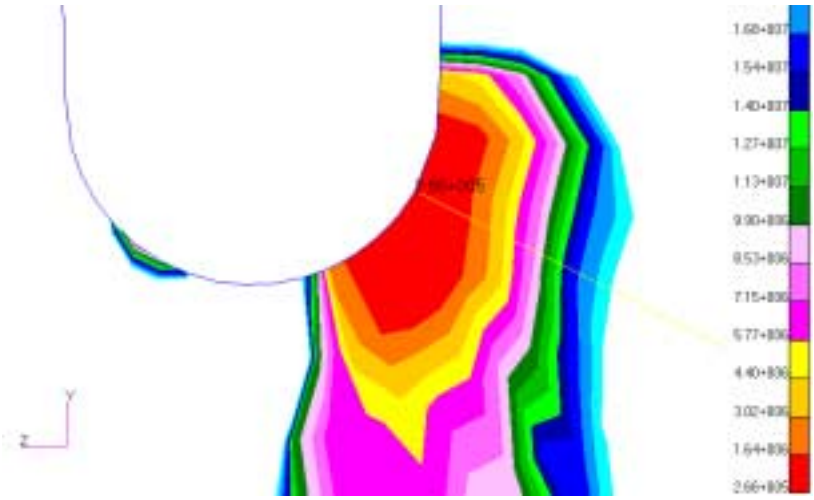


Figure 27. Plot of total life zoomed 2

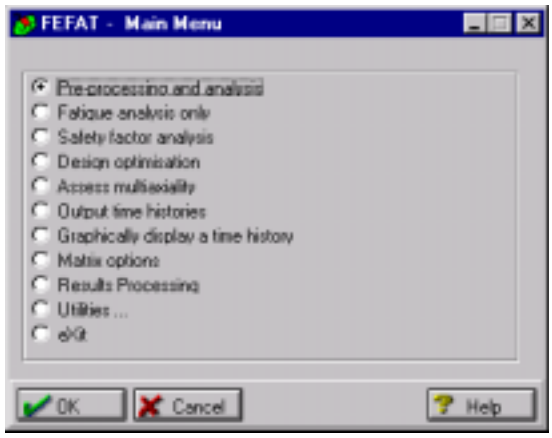


Figure 28. The form spawned from Job Control → Interactive

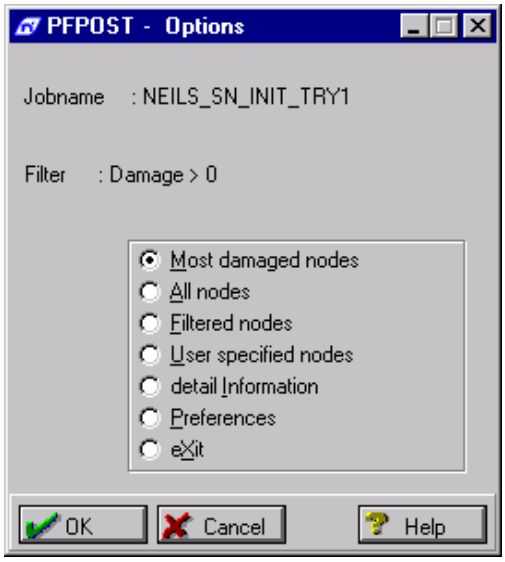
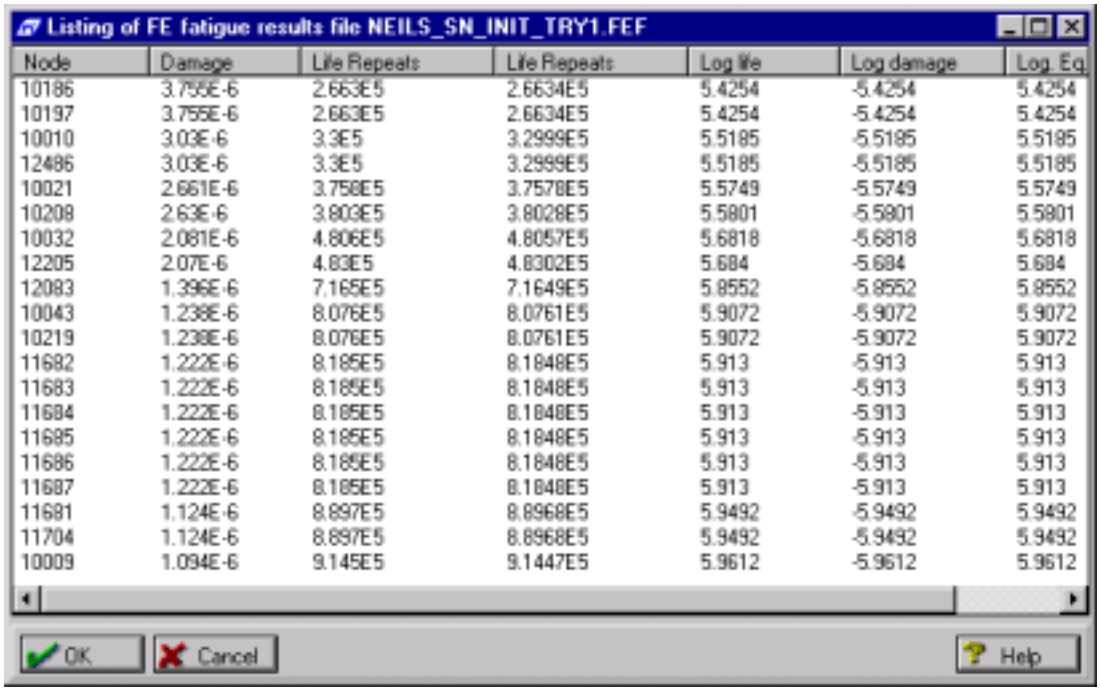


Figure 29. The form spawned by the "Results Processing" button



Node	Damage	Life Repeats	Life Repeats	Log life	Log damage	Log Eq
10186	3.755E-6	2.663E5	2.6634E5	5.4254	-5.4254	5.4254
10197	3.755E-6	2.663E5	2.6634E5	5.4254	-5.4254	5.4254
10010	3.03E-6	3.3E5	3.2999E5	5.5185	-5.5185	5.5185
12486	3.03E-6	3.3E5	3.2999E5	5.5185	-5.5185	5.5185
10021	2.661E-6	3.758E5	3.7578E5	5.5749	-5.5749	5.5749
10208	2.63E-6	3.803E5	3.8028E5	5.5801	-5.5801	5.5801
10032	2.081E-6	4.806E5	4.8057E5	5.6818	-5.6818	5.6818
12205	2.07E-6	4.83E5	4.8302E5	5.684	-5.684	5.684
12083	1.396E-6	7.165E5	7.1649E5	5.8552	-5.8552	5.8552
10043	1.238E-6	8.076E5	8.0761E5	5.9072	-5.9072	5.9072
10219	1.238E-6	8.076E5	8.0761E5	5.9072	-5.9072	5.9072
11682	1.222E-6	8.185E5	8.1848E5	5.913	-5.913	5.913
11683	1.222E-6	8.185E5	8.1848E5	5.913	-5.913	5.913
11684	1.222E-6	8.185E5	8.1848E5	5.913	-5.913	5.913
11685	1.222E-6	8.185E5	8.1848E5	5.913	-5.913	5.913
11686	1.222E-6	8.185E5	8.1848E5	5.913	-5.913	5.913
11687	1.222E-6	8.185E5	8.1848E5	5.913	-5.913	5.913
11681	1.124E-6	8.897E5	8.8968E5	5.9492	-5.9492	5.9492
11704	1.124E-6	8.897E5	8.8968E5	5.9492	-5.9492	5.9492
10009	1.094E-6	9.145E5	9.1447E5	5.9612	-5.9612	5.9612

Figure 30. Table of fatigue life results for MVision 7050 T74 data

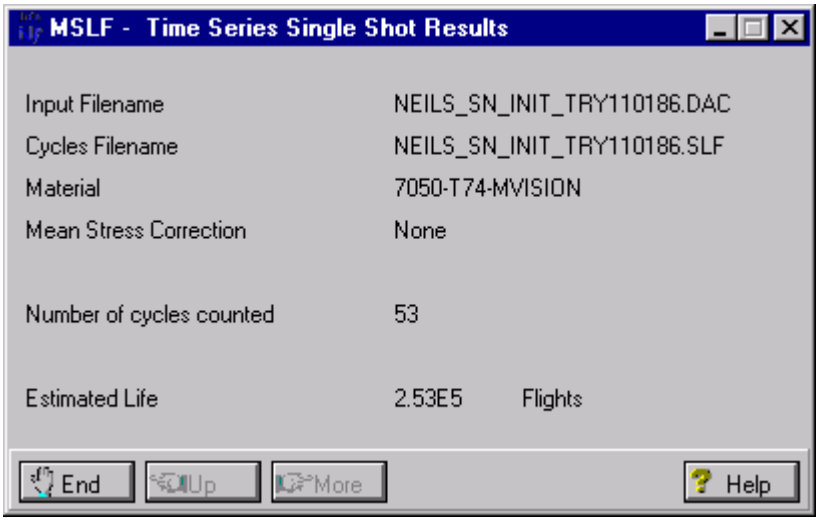


Figure 31. Fatigue life calculated with MVision 7050 T74 data

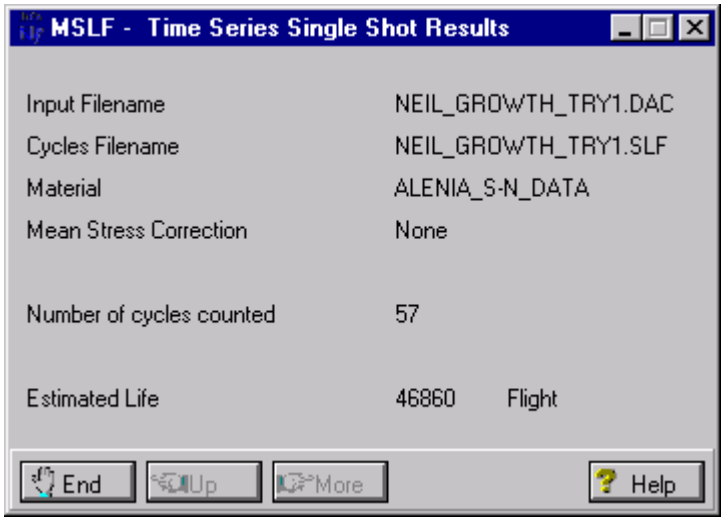


Figure 32. Fatigue life calculated with Source no1 7050 T74(Kt=3.1) data and reference stress time signal

Sensitivity to Loading Level

In order to assess the influence of load (stress) magnitude the ***Design Optimisation*** form can be run by choosing ***Job Control → Interactive → Design Optimisation*** buttons (see **Figure 33**). From this form a sensitivity analysis can be run for varying load levels. For instance (1.0,2.5,0.1) returns results from 1.0 to 2.5 in steps of 0.1. The result of this is shown in **Figure 34**. So, for instance, the fatigue life for a load (stress) increase of 1.7 is 6600 flights. The results are summarized in Table 1.

Table 2. Summary of Stress-Life results

S-N data used	***Fatigue life in flights***
Mvision 7050 T74 data	253E3
Source no1 7050 T74 data Kt=1.0	174E3
Source no1 7050 T74data Kt=3.1 plus reference stress	46E3
Mvision 7050 T74 data with stresses increased by 1.7	6.6E3

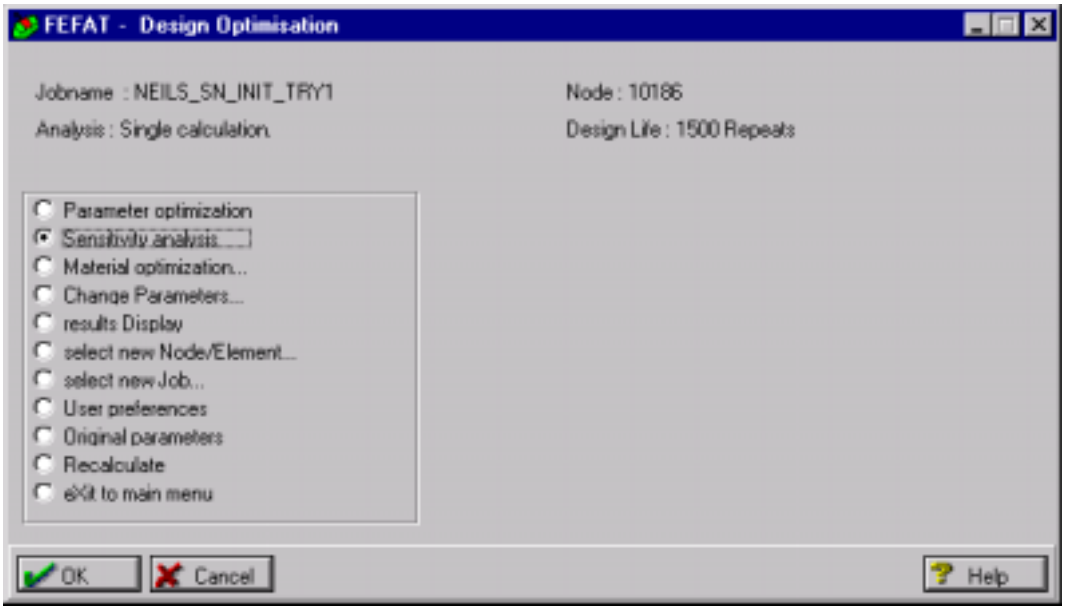


Figure 33. The Design Optimisation form

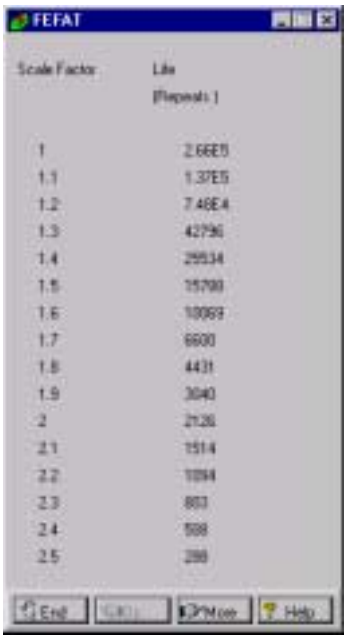


Figure 34. Load levels varied from 1.0 to 2.5

Investigation of Stress Conditions – Marc & Nastran

The stress conditions in the region of the crack were investigated with both Marc and Nastran. Based on these results it was concluded that principal stresses, rather than shear stresses, were driving crack behaviour. Typical results are shown in Appendix 2 and Appendix 3.

Stress Distribution Across Critical Crack Plane

Appendix 4 shows the stress distributions across the critical crack direction. These results show some very strange behaviour in the region of the panel stiffener and should be investigated further.

Investigation of Compliance Function with Marc

A limited amount of work was done with Marc on a 2D model to generate K and hence a Compliance Function Y for the panel. The results are shown in blue in **Figure 35**. Again this shows very strange behaviour in the region of the stiffener and it is recommended that this be investigated further with a 3D model. The comparison curve generated with the MSC.Fatigue Compliance Function generator (see later) is shown in red.

In addition some work was done to check the possibility of the crack diverting away from the pre determined directions. In order to check this the J Integral value was plotted against divergence angle and this confirmed that the "least line of resistance" at a crack length of 8mm was straight ahead (see Figure 36).

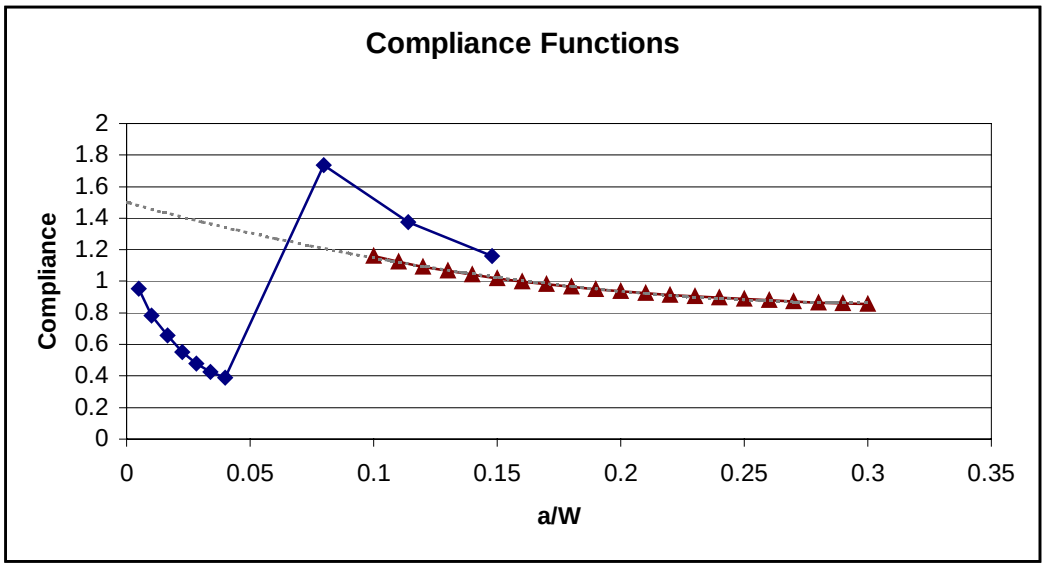


Figure 35. Blue curve represents result from Marc normalized by 55Mpa
Red curve is from MSC.Fatigue for an edge crack in center hole, R=15 and W =245.

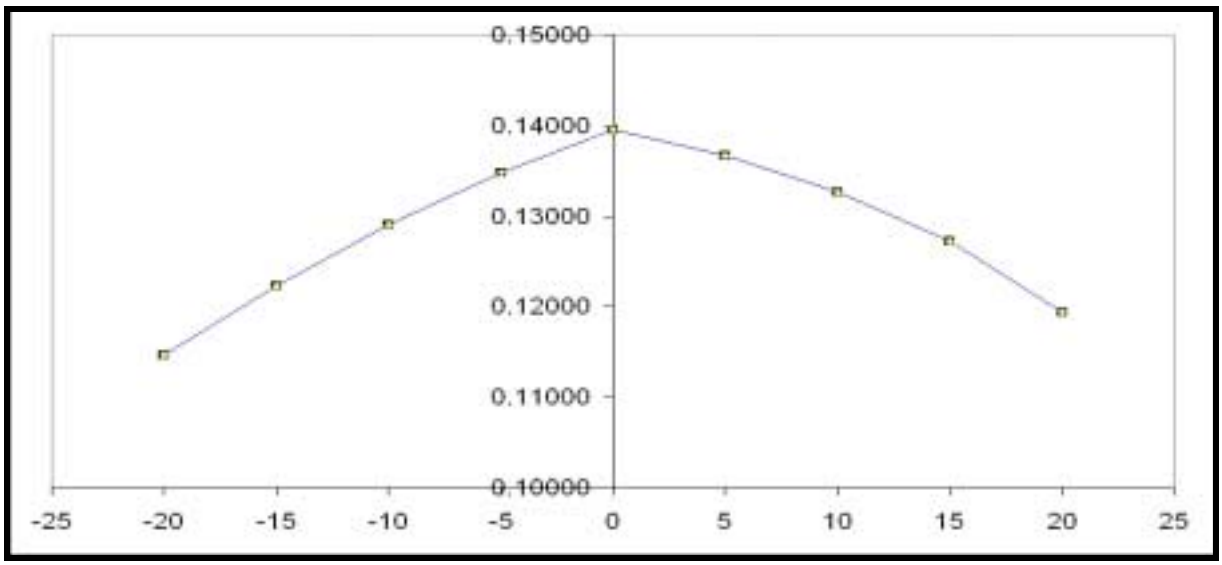


Figure 36. The J Integral value plotted against crack direction

Crack Growth Calculation With MSC.Fatigue

The Concept of Stress Intensity

Modern calculations use the concepts of Fracture Mechanics. These were originally developed to compute the static loads that can be carried by components containing cracks, before being extended to cover fatigue crack propagation. The difficulties can be illustrated by referring to Figure 37. A circular hole in a plate carrying uni-axial tension has a K_T of 3, which increases if the hole is elongated in a direction perpendicular to the principal stress. A possible model of a crack is an ellipse like this but with finite length *a* and zero width *b*. The simple formula given in the figure then predicts that K_T will be infinite, so that any value of the nominal stress σ will cause an infinitely high tension at the crack tip. This implies that the crack will propagate catastrophically by tearing. Experiments show that this does not happen. Cracked bodies will carry some load, so the analysis must be faulty and an alternative is needed.

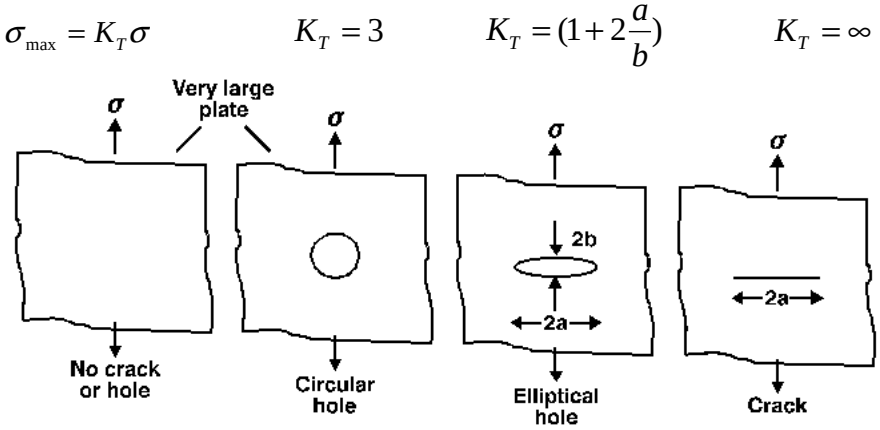


Figure 37. Elastic stress concentrations

Early theories concentrated on the energy absorbed when the crack moves forward, but we now concentrate on the stress field around the tip of the crack. Westergaard showed that this field can be expressed in such a way that all terms (normal stresses and shear stresses on all elements, Figure 38) can have a common factor $\sigma(\pi a)^{1/2}$ extracted. Geometry, form of loading, and the way the crack will extend also have effects that it may be possible to quantify for the whole field. Combining these into a factor *Y* we can identify a parameter that characterises the whole stress field near to the crack tip. Calling this the ***crack tip stress intensity factor***, *K*, we have:-

$$K = Y\sigma(\pi a)^{1/2}$$

If the crack extends when *K* reaches a critical value we will have a criterion for determining the stress σ which will extend a crack of length *a* when *Y* has a certain value. Tests show that this is the case.

Continuing to limit the discussion to the case of static loading, the critical value that K must reach is a material property called ***fracture toughness***, symbol K_{Ic}. Analysis is often limited to the case where the plastic zone near to the crack tip is small (Figure 39), and the topic is then called Linear Elastic Fracture Mechanics (LEFM). The three modes of extension that can take place are identified as Mode I, Mode II and Mode III as shown in Figure 40 and solutions for *Y* will depend on the mode as well as on component geometry. Closed-form solutions for *Y* exist in many cases, and numerical solutions using FEA are common. Several

compendia of expressions have been published such as the well known Rooke and Cartwright publication. Most of these are included in the MSC.Fatigue database.

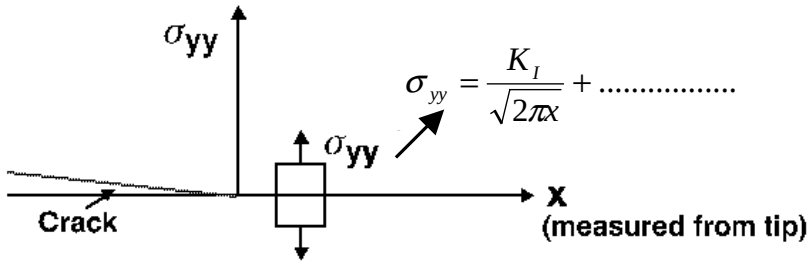


Figure 38. The concept of stress intensity

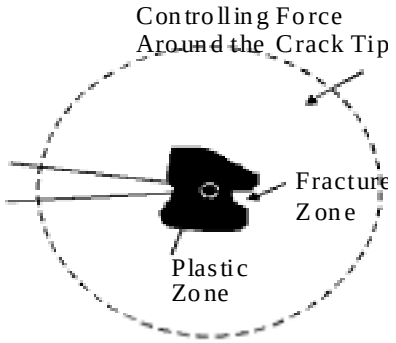


Figure 39. Crack tip yielding

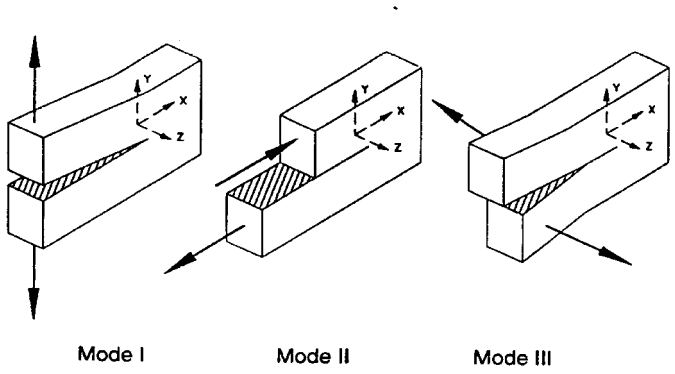


Figure 40. The 3 modes of cracking used in fracture mechanics

Fatigue Crack-Propagation and LEFM

Static LEFM provides a numerate criterion for the onset of catastrophic failure in a fatigue test. If a crack is propagating under repeated loading *a* will steadily increase and the value of *K* when the dynamic load is a maximum will also increase. When this value, K_{Max}, reaches K_{Ic} , a material property, the crack will extend catastrophically. LEFM contributes more than this to fatigue, though. In a constant amplitude fatigue test ranging from σ_{Max} to σ_{Min} values of *K* can be computed for the extremes of load, say K_{Max} and K_{Min}. The difference between these is the range of crack tip intensity factor, say ΔK. As the rate of crack propagation d*a*/dN depends on ΔK its value can be calculated. The rate will depend on σ and on crack length *a*, which will increase as the crack extends. To carry out tests at constant ΔK crack length must be monitored and the range of load decreased in proportion to $a^{1/2}$.

Figure 41 shows typical results for tests carried out at constant ΔK. There are three regions. Over the middle range of ΔK the relationship is linear. Both scales are logarithmic, which means that an expression for this middle portion is:

$$da/dN = C(\Delta K)^m$$

where da/dN = current rate of crack propagation,
a = current crack length, and C, m are material properties.

This is the most-used expression, called the ***Paris-Erdogan*** equation. For high values of ΔK the graph diverges from a straight line. This is not particularly important in most practical cases because the crack is growing so rapidly that big variations in the formula used here make little difference to the predicted life. Much more important is the deviation from the straight line at low ΔK values. If the line becomes vertical, as it does in many reported data, there is a ***threshold*** value of ΔK below which the crack growth rate is zero.

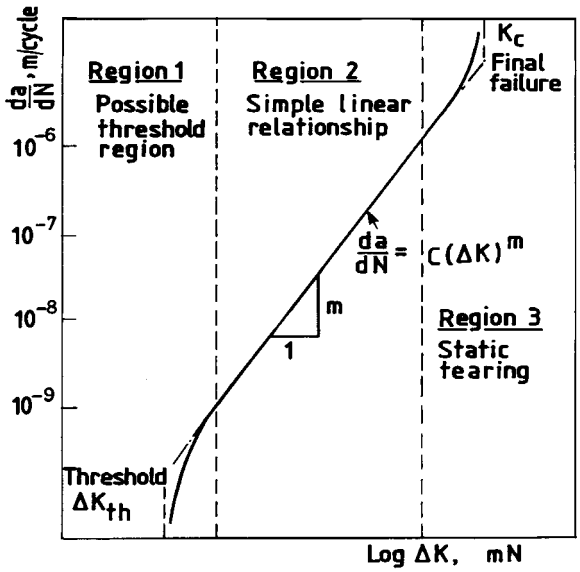


Figure 41. Crack propagation rates at constant ΔK

The majority of fatigue crack propagation studies have been associated with long cracks (i.e. of order 10 mm) and low cyclic stresses, which are well described by linear elastic fracture mechanics. Since Mode I cracking is the most easily studied, using standard specimen geometries, Mode II and Mode III crack growth data are scarce. Mode I is the most common type of in-service failure.

The curve in Figure 41 is sometimes called the ***apparent*** ΔK curve. There are many effects that the Paris equation does not take into account, such as crack closure, corrosive environments, the influences of a notch, and static fracture mode contributions. One way to model these effects is to derive an ***effective*** ΔK curve modifying the apparent ΔK curve through all three of its regions. This effective ΔK is then used in the Paris equation to determine crack growth

Stresss Intensity Factor *K* Versus Compliance-Function *Y*

The factor *Y* is sometimes called the ***Compliance-Function***. In physical terms it is the change in stiffness or flexibility as the crack grows, i.e., the structure becomes more compliant as the crack gets longer. *Y* is unity for a central through-thickness crack in an infinite body and 1.12 for an edge crack. For more realistic geometries, the Compliance-Function, *Y*, is normally more complex. Many hundreds of *K* solutions are available. The designer's task is not to derive new expressions for *Y* but to identify the geometry and loading present and find an existing formula.

The relationship between stress intensity, stress, and crack length is known as the ***fracture mechanics triangle*** (Figure 42). If two of the corners are known the other can be derived.

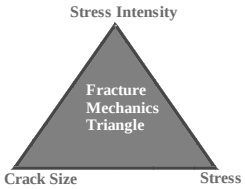


Figure 42. The fracture mechanics triangle

The fracture mechanics triangle (Figure 42) allows a designer to compute any one of the three parameters given the other two. Using ΔK and the Paris equation the instantaneous rate of crack propagation at a given value of *a* can then be calculated. An extension of this, using some form of integration, will give the number of cycles needed to propagate a crack from an initial to a final length. This leads to the fatigue crack propagation rectangle of Figure 43, where knowing any three corners allows the fourth to be computed. When considering crack growth, then, there is a relationship between stress range and life just as there is with the Stress-Life (S-N) method. In crack growth, though, life is closely related to initial and final crack lengths. This forms the basis of a life estimation method that underlies the Damage-Tolerance philosophy.

Figure 43. The fatigue crack propagation rectangle

What is the Meaning of "Nominal Stress" as Used With a Compliance-Function in FE Based Crack Growth Calculations?

The definition of ***nominal*** stress, for crack growth calculations, is similar to that used in component S-N curves. Rather than try to consider the stress in the region of the crack tip, the stress conditions around the crack tip are characterised using *K*, the stress intensity factor. This value is a function of the nominal stress (from the uncracked body) and the crack length. The value of *K* is further modified by the *K*-solution, or Compliance-Function, this is a function of crack ratio. The crack ratio varies from 0 (uncracked) to 1 (fully cracked). This Compliance takes into account two effects. Firstly as the crack grows the stresses may re-

distribute and secondly the crack may be growing into higher or lower-stressed areas. So it is very important to use the correct "nominal" stress.

Using Fracture Mechanics in Damage-Tolerant Design.

In one version of Damage-Tolerant design the initial crack length a_i is determined by the longest crack likely to be missed at inspection, and the final crack length a_f is the one which would cause catastrophic failure. If the applied fatigue loading is constant-amplitude, the steps in the calculation are:

(i) Estimate the size of crack likely to be present in the component when it is first put into service, a_i.

(ii) From a measured value of K_{Ic}, estimate the maximum crack length, a_f, which the component will tolerate when the applied stress reaches maximum tension. An expression for the crack-tip stress intensity factor will be needed.

(iii) Using the same expression for crack-tip stress intensity factor, calculate ΔK.
(iv) Substitute ΔK into the Paris equation to obtain a crack propagation rate. This will put da/dN in terms of crack length a.

(v) Integrate this equation between $a = a_i$ and $a = a_f$ to give the number of cycles needed to grow a crack from a_i to a_f. This is the predicted life of the component. A classical integration is adequate if the Compliance Function is constant. Other Compliance Functions will require a numerical integration, as will any non-constant amplitude signal.

Implementation in MSC.Fatigue

Again the starting point for an MSC.Fatigue analysis is the MSC.Fatigue form spawned from the ***tools*** top bar menu as shown in Figure 44. Now, because Crack Growth has been chosen there is a new button ***Compliance Function*** at the top of the form. This allows ***PKSOL*** to be accessed which is a useful tool for generating Compliance Functions from a standard library, for entering externally defined CF's, and also for solving the Fracture Mechanics Triangle to obtain, for instance, K from crack length and stress level. Examples of this are shown in Figure 45 to Figure 51.

Once a Compliance Function is available the rest of the MSC.Fatigue forms can be generated in a very similar way to the S-N approach. Figure 52 shows the ***Materials*** form. In this form the appropriate da/dN curve is specified. da/dN data is somewhat more complicated than S-N data, allowing a number of modifications to be made to deal with, for instance, history effects, crack closure, etc. These were all turned off in the analysis so that a standard Paris type crack growth calculation was performed. In order to ensure that all these effects are turned off the values for D0 and D1 in the form (see Figure 53) have to be set to very low values, say 0.1 (the reader is referred to the MSC.Fatigue User Manual for more details). The one major difference in the Material form is that the region defined here is the region from which the reference stress is to be calculated.

Figure 54 to Figure 58 show the various da/dN curves used.

In order to perform a crack growth calculation an appropriate reference stress has to be calculated form the model. This should be representative of the "nominal" stress in the model

if the feature being modelled (hole) was not there. In order to assess this the stress distributions shown in Appendix 4 were carefully scrutinised and the element area shown in Figure 59 was subsequently chosen in order to calculate an average “reference” stress. The stress history calculated form this region is shown in Figure 60.

In the usual way Figure 61 and Figure 62 show the Loading form. Figure 63 and Figure 64 show typical final results for the critical location (node 10186). Finally, in order to assess the effect of corrosion, the NASGRO data for 7050 T74 corrosion (supplied on the MSC.Fatigue CD) was used to assess the effect on crack growth rates. These results, and all other results, are summarised in Table 3. These results clearly show the detrimental effects of corrosion and increased load level.

Table 3. Summary of crack growth results for different da/dN data

Da/dN data	***Fatigue Life (flights)***
7050 T73652 LT – original stress	67E3
7050 T73652 LT – stress x 1.7	10.7E3
7050 T73652 TL – original stress	207E3
7050 T73652 TL – stress x 1.7	28E3
NASGRO 7050 T74 in air – original stress	133E3
NASGRO 7050 T74 in air – stress x 1.7	15E3
NASGRO 7050 T74 in Humid Air	30E3
NASGRO 7050 T74 in Salt Fog	20E3

Figure 44. The MSC.Fatigue crack growth form

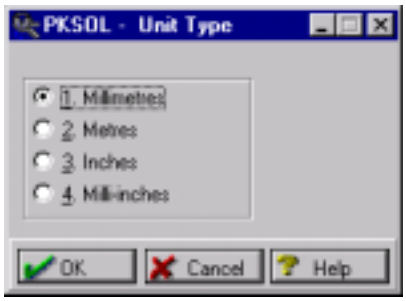


Figure 45. Using PKSOL to generate a Compliance Function

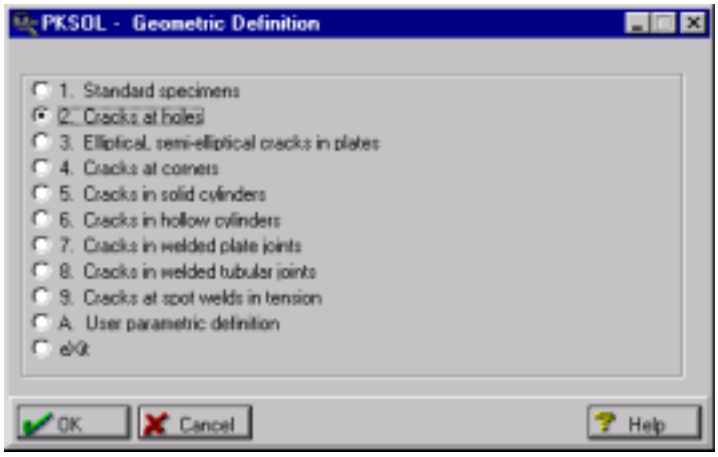


Figure 46. Standard configurations available for computing a Compliance Function

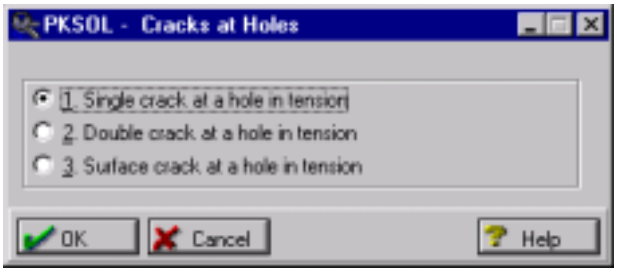

PKSOL - Cracks at Holes
1. Single crack at a hole in tension
2. Double crack at a hole in tension
3. Surface crack at a hole in tension
OK
Cancel
Help

Figure 47. Choose single crack at hole in tension

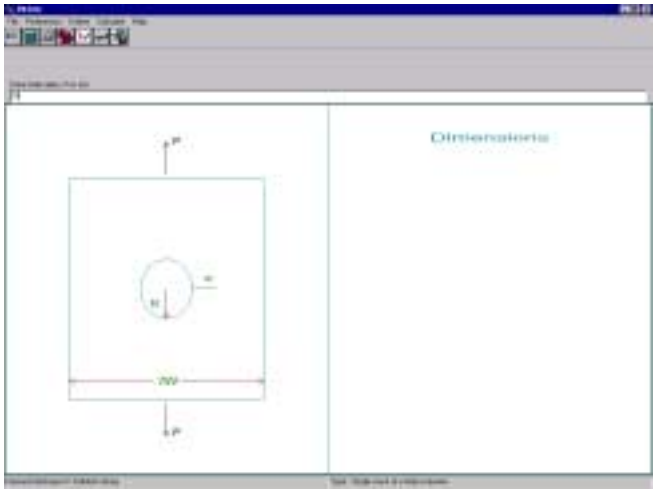


Figure 48. Graphical plot of single crack at hole in tension

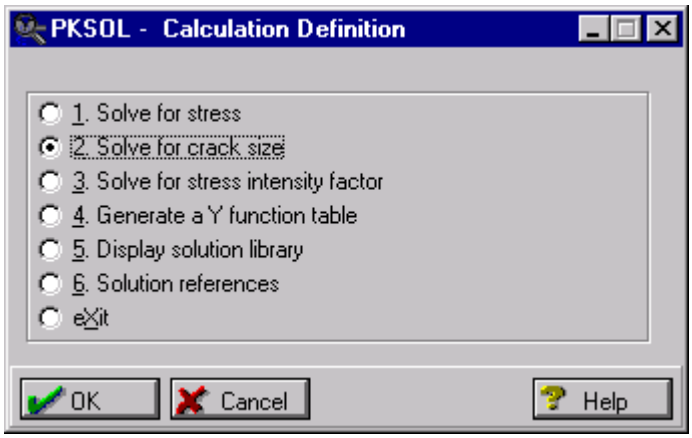


Figure 49. Options for solving the Fracture Mechanics Triangle

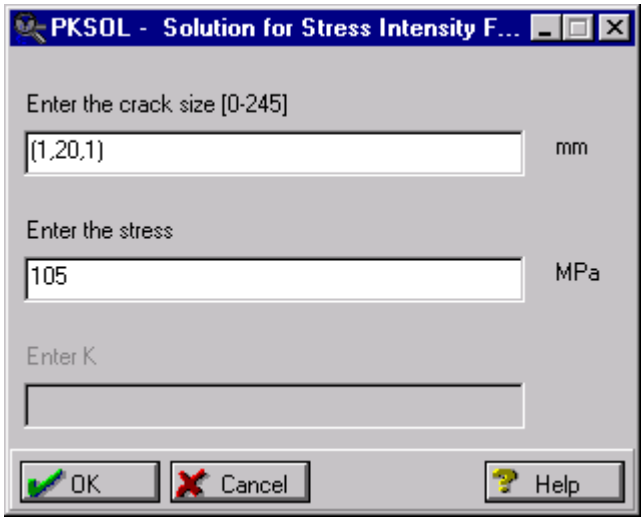


Figure 50. Solving the Fracture Mechanics Triangle for K

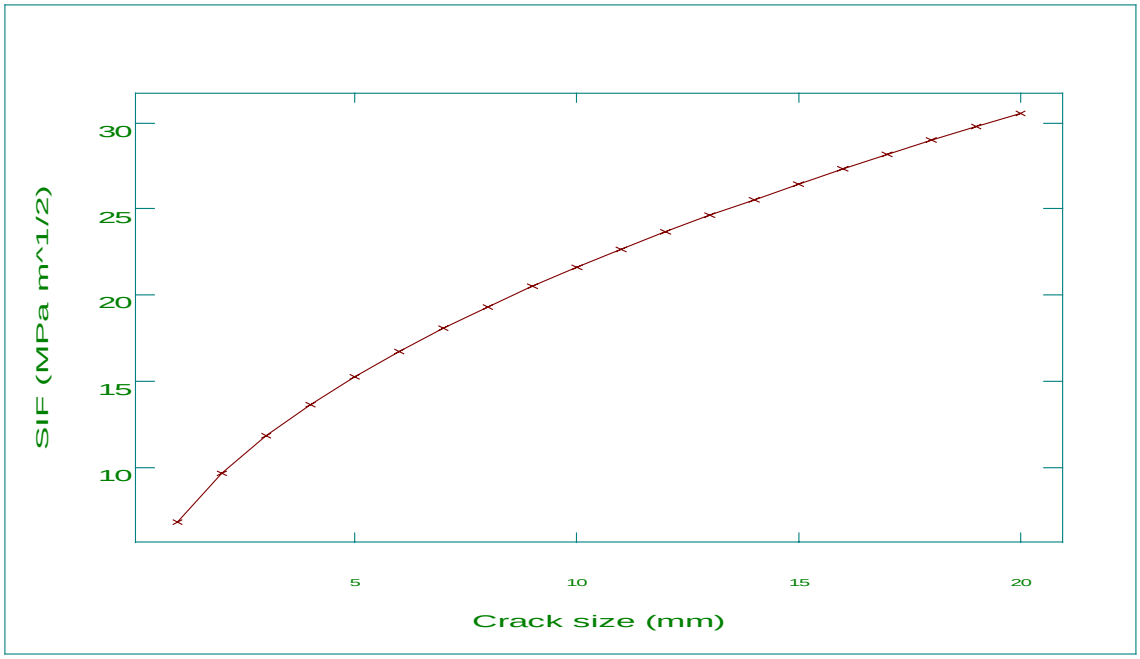


Figure 51. A typical result showing K versus crack size.

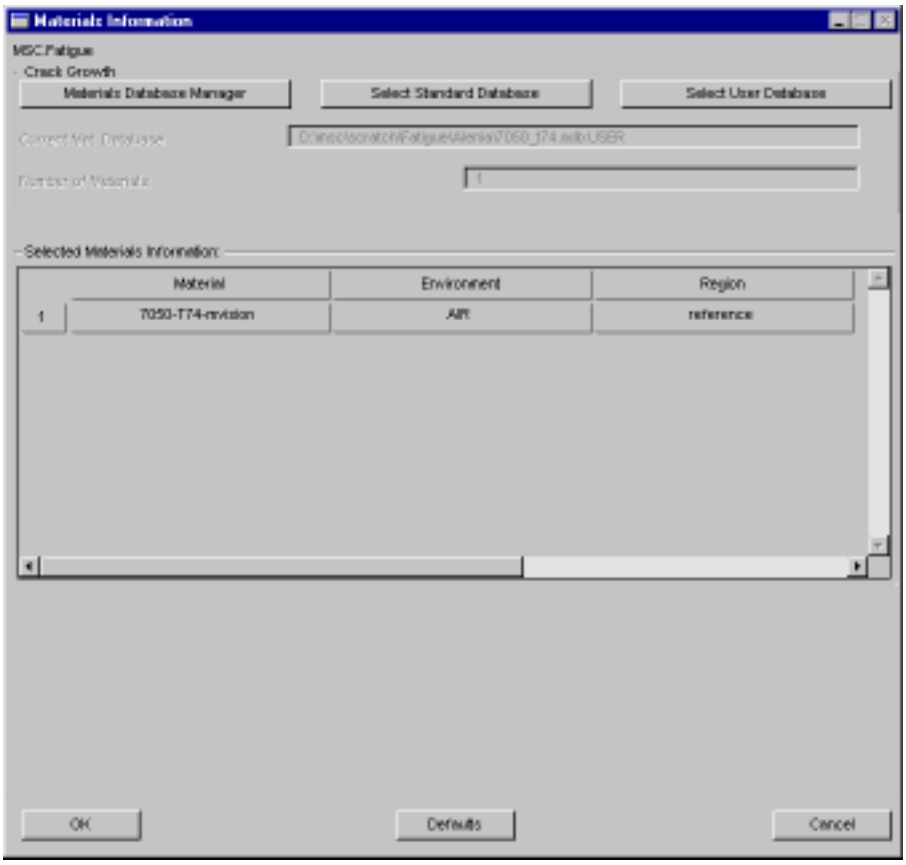


Figure 52. The materials form in Fracture

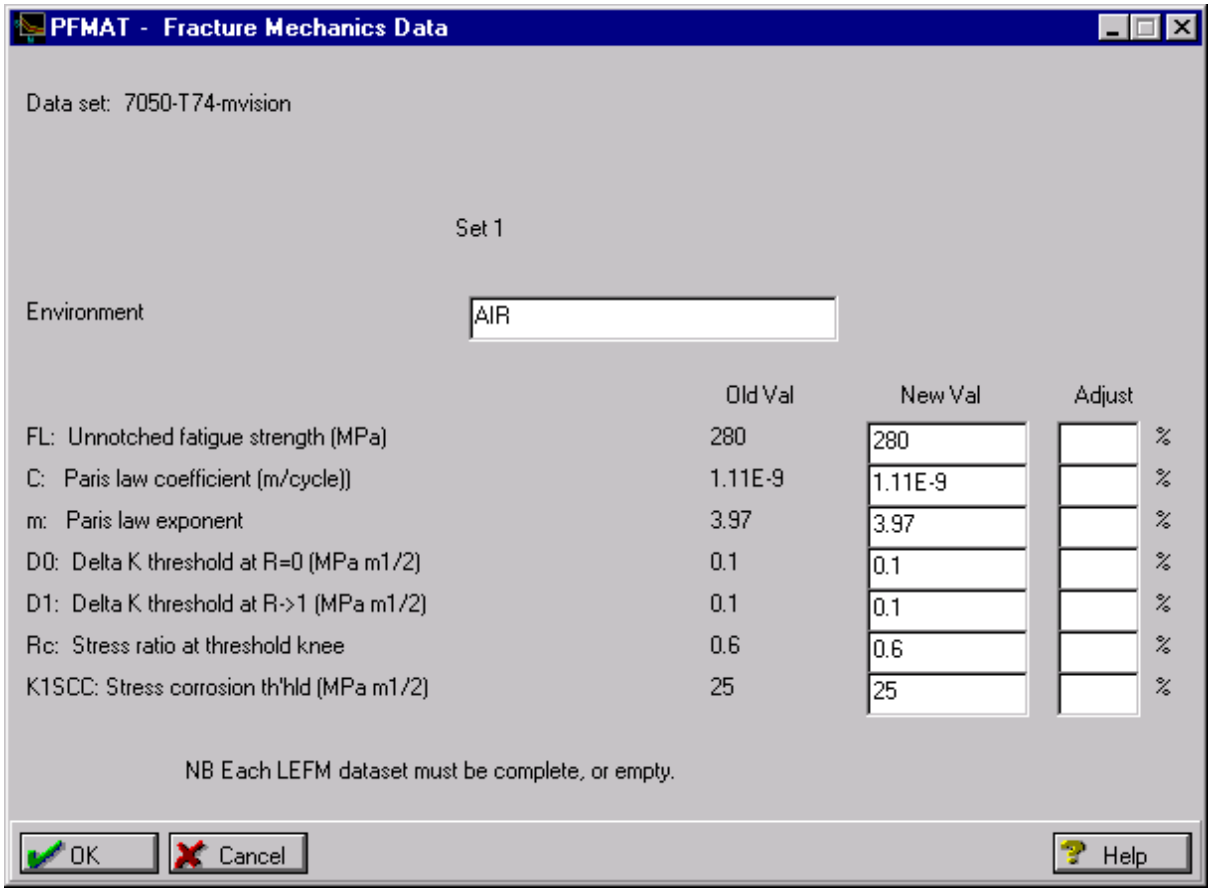


Figure 53. Editing the Materials form

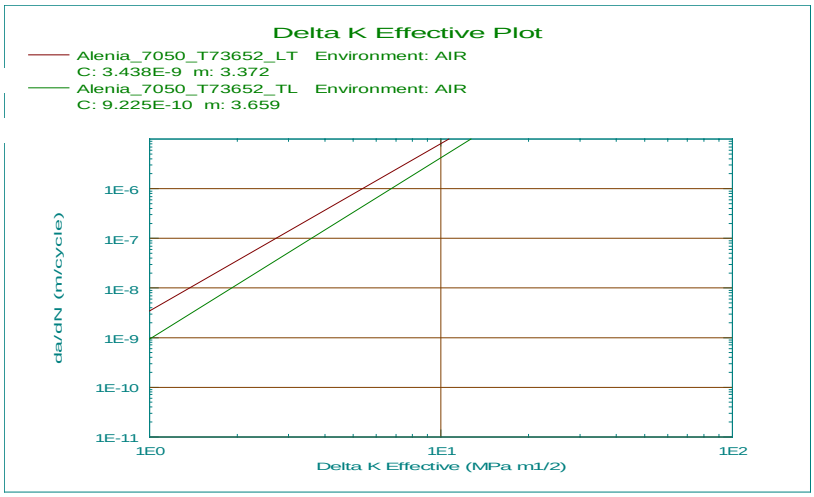


Figure 54. A comparison of 7050 T73652 da/dN souce no1 data in LT and TL directions

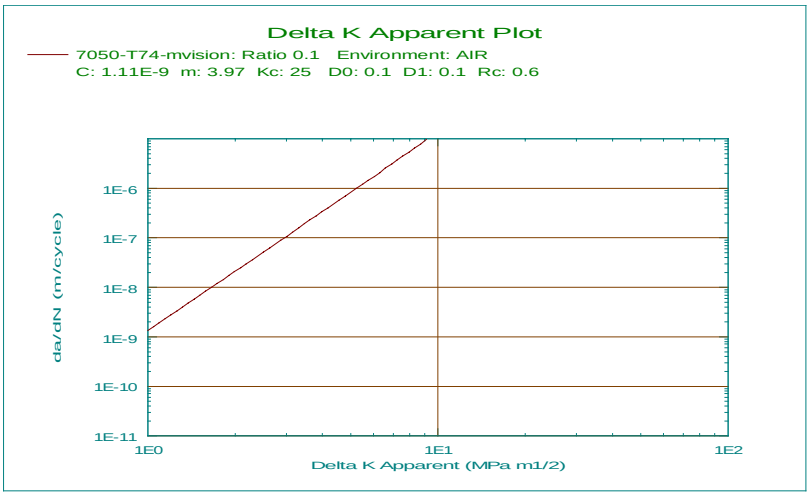


Figure 55. NASGRO 7050 T74 Apparent Delta K da/DN data

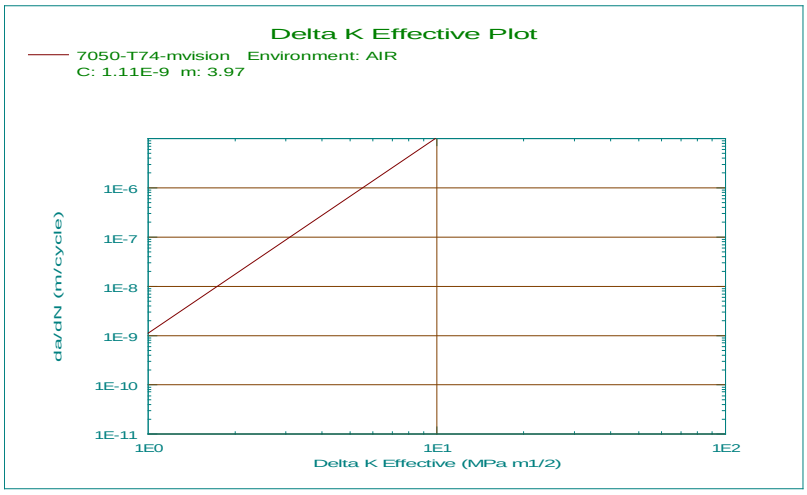


Figure 56. NASGRO 7050 T74 Effective Delta K da/DN data

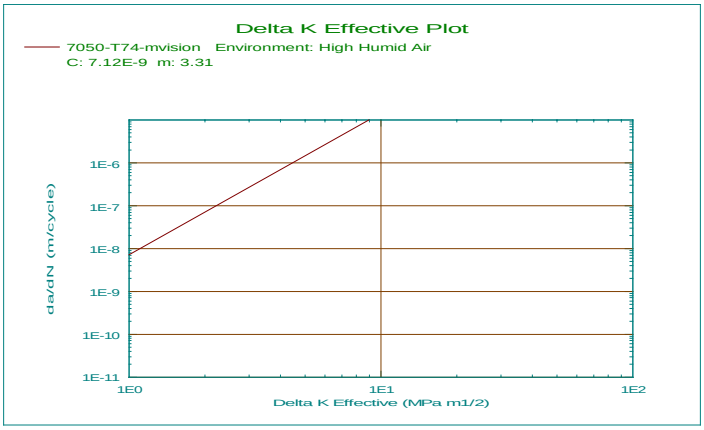


Figure 57. NASGRO 7050 T74 data for High Humidity Air

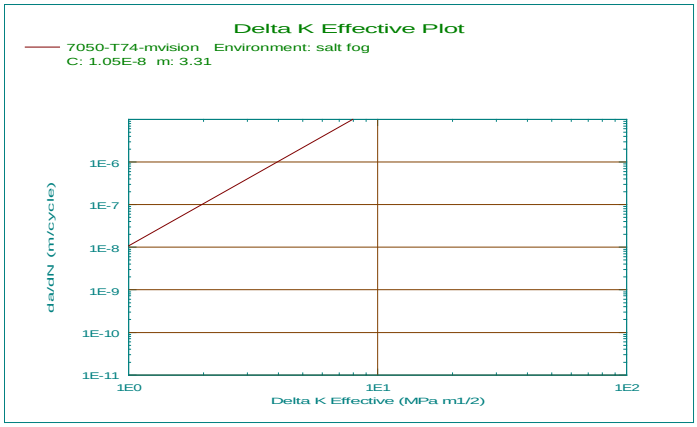


Figure 58. NASGRO 7050 T74 data for Salt Fog

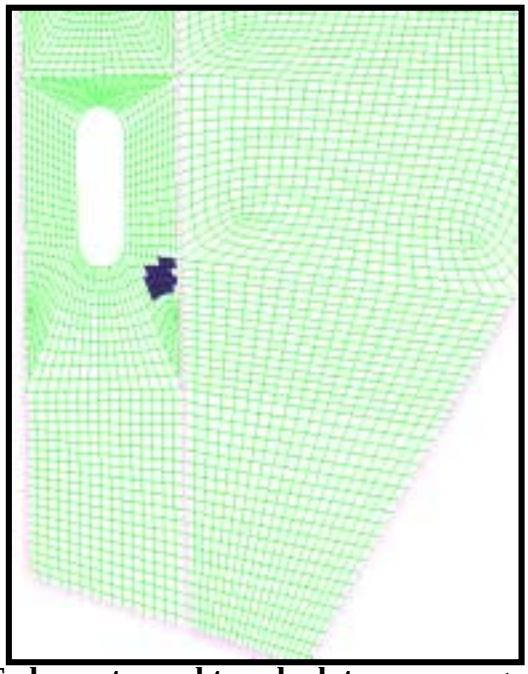


Figure 59. The FE elements used to calculate an average "reference" stress

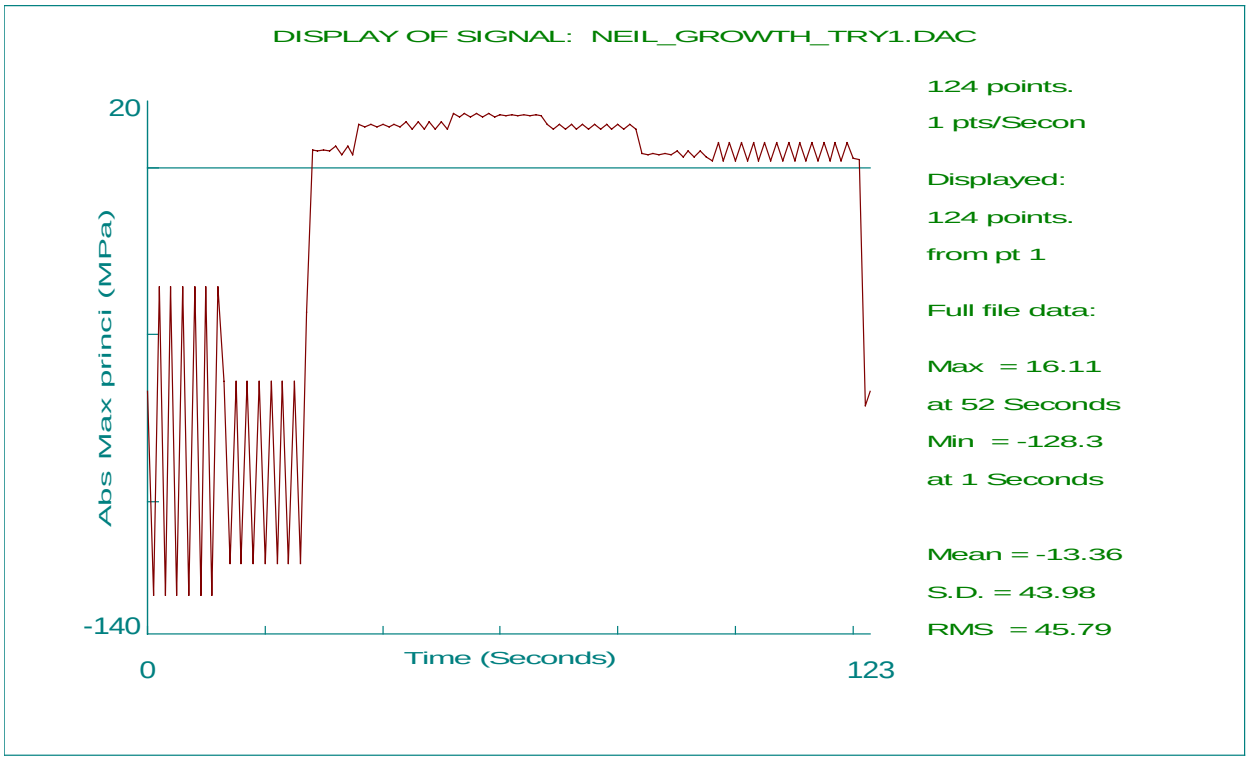


Figure 60. The reference stress calculated from the elements shown in Figure 59**.**

Figure 61. The Loading form

Figure 62. The Loading form showing the same 12.5 scale factor

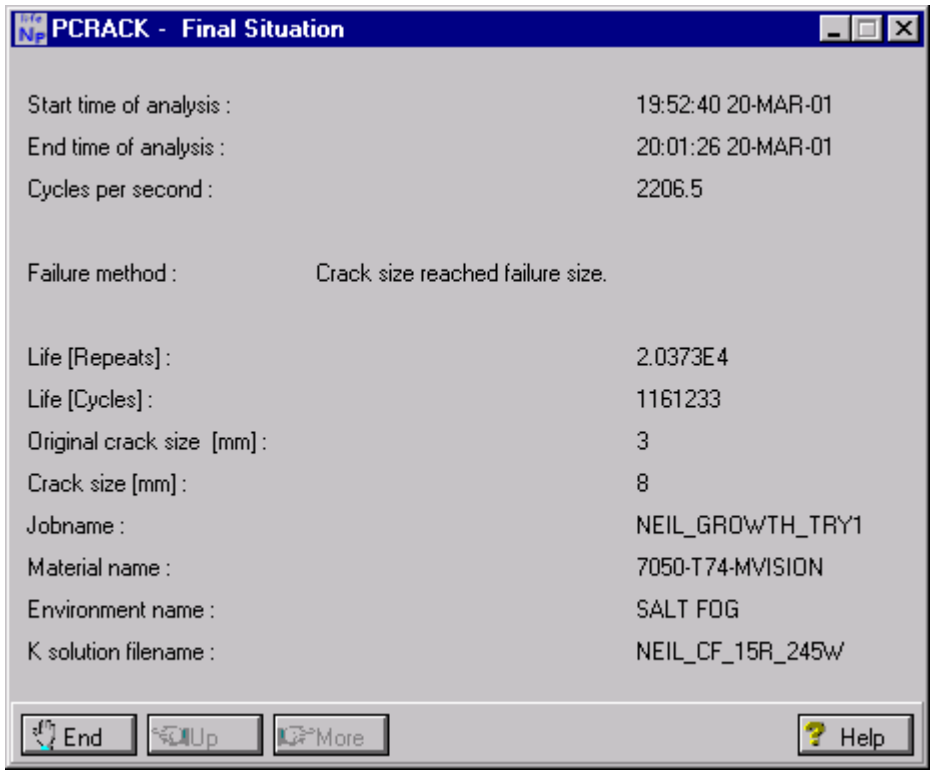

PCRACK - Final Situation
Start time of analysis : 19:52:40 20-MAR-01
End time of analysis : 20:01:26 20-MAR-01
Cycles per second : 2206.5
Failure method : Crack size reached failure size.
Life [Repeats] : 2.0373E4
Life [Cycles] : 1161233
Original crack size [mm] : 3
Crack size [mm] : 8
Jobname : NEIL_GROWTH_TRY1
Material name : 7050-T74-MVISION
Environment name : SALT FOG
K solution filename : NEIL_CF_15R_245W
End
Up
More
Help

Figure 63. An example of a final result from a crack growth analysis

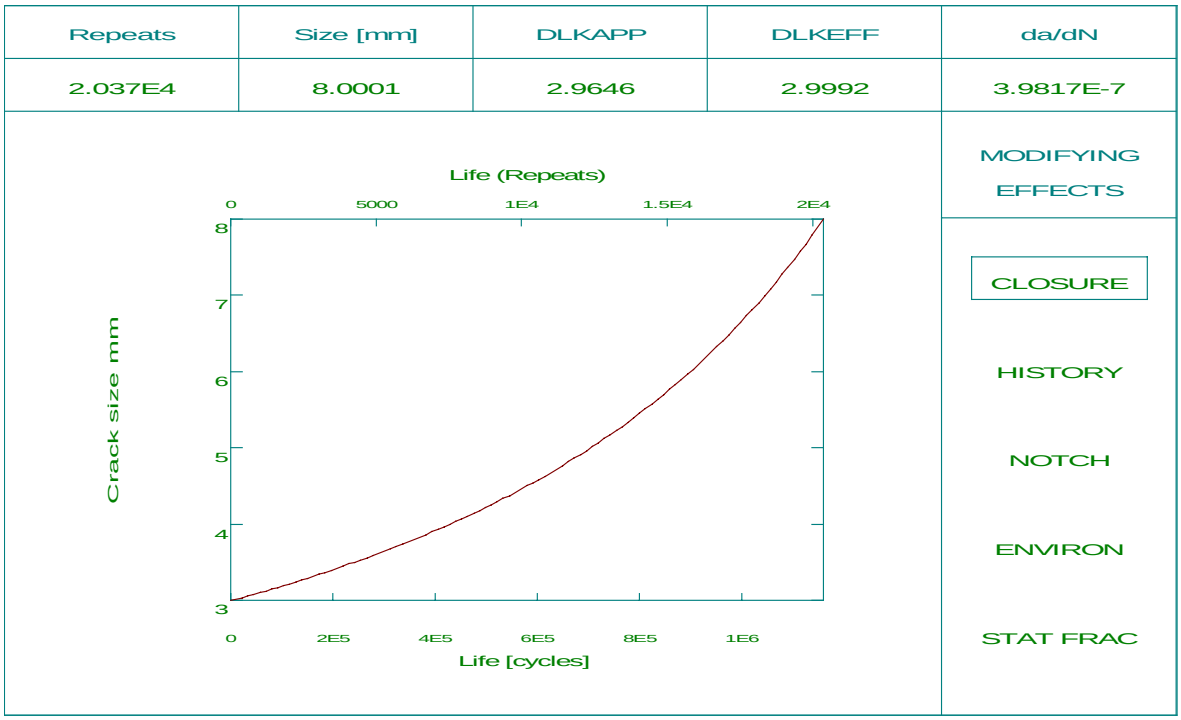


Figure 64. A typical da/DN final crack growth curve

Conclusions and Recommendations For Future Work.

Stress-Life and crack growth calculations have been performed on the main frame panel shown in **Figure 2**. Various sensitivity studies have been performed which show the effect on fatigue life of, for instance, increased loading levels.

Appendix 1. S-N and Crack Growth data supplied by Source no1

Source no1 was the material Databank of Alenia Aerospazio Divisione Aeronautica, Pomigliano D'Arco Napoli.

FATIGUE DATA

MAT:	ALUMINIUM
ALLOY AND TEMPER:	7050-T74
FORM:	DIE FORGING
GEOMETRY:	OPEN HOLE (DIA = 4.8 mm; Kt = 3.1)
HOLE CONDITION:	DRILLED AND DEBURRED
THICKNESS / WIDTH:	8 mm
FATIGUE CYCLE:	CONSTANT AMPLITUDE; R = 0.06
ROUGHNESS:	3.2µm
GRAIN DIRECTION:	LONGITUDINAL
CHEMICAL TREATMENT:	CHROMIC ACID ANODIZE

Experimental S-N Curve equation (mean value):

$$N = 10^5 \left(\frac{600 - S\max(1+R)}{S\max(1-R)} \quad \frac{40.2}{200 - 40.2} \right)^{3.6}$$

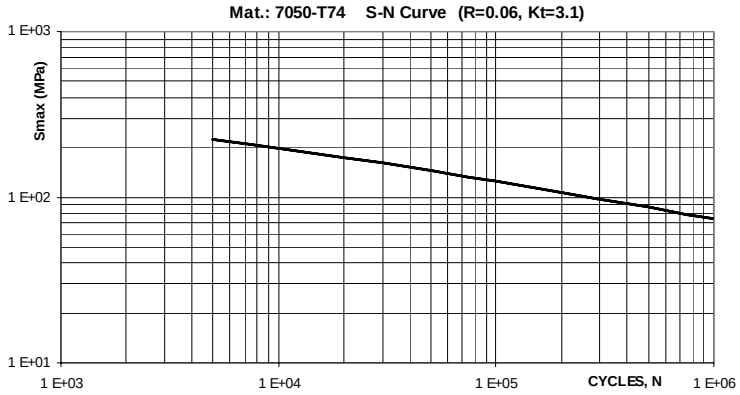
CRACK PROPAGATION DATA (Forman)

7050-T73652 (Forg.) - Tk = 6 mm, grain dir: LT

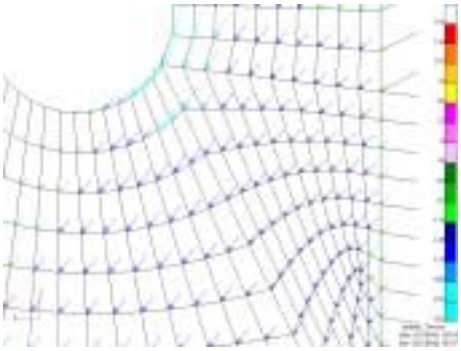
	Units of measurement	
	m, Mpa, Mpa √m	Inches, Ksi, Ksi√in
m	3.372	3.372
C	3.438485E-09	1.692889E-07
Kc	70.98972	64.60432

7050-T73652 (Forg.) - Tk = 6 mm, grain dir: TL

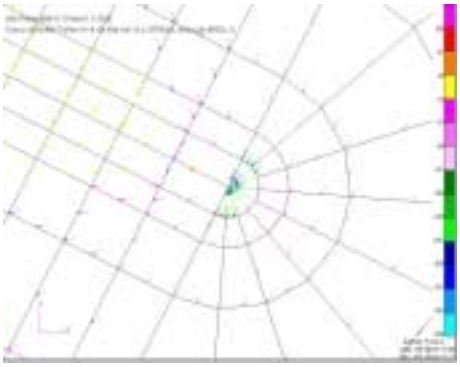
	Units of measurement	
	m, Mpa, Mpa √m	Inches, Ksi, Ksi√in
m	3.659	3.659
C	9.224681E-10	1.692889E-07
Kc	40.98114	37.29496



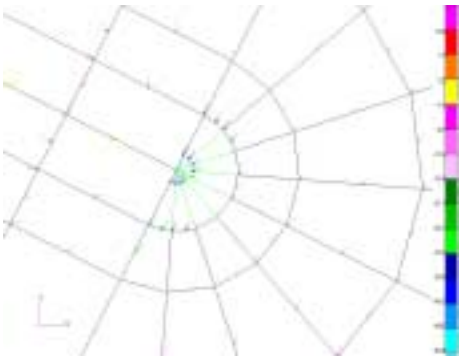
Appendix 2 Marc Results



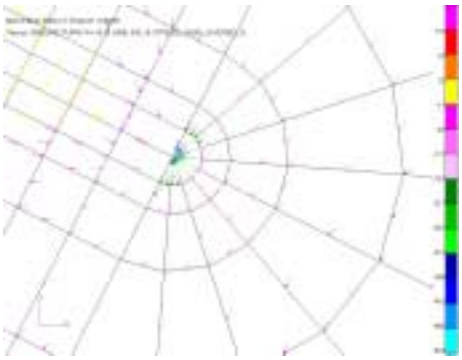
Min Principal a=0



Min Principal a=1

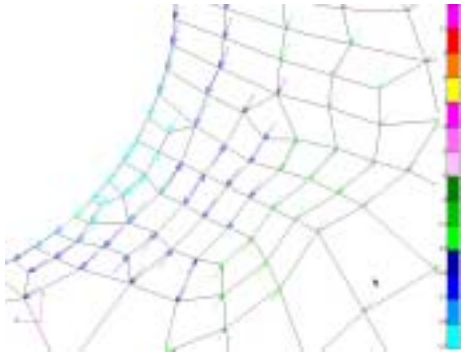


Min Principal a=2

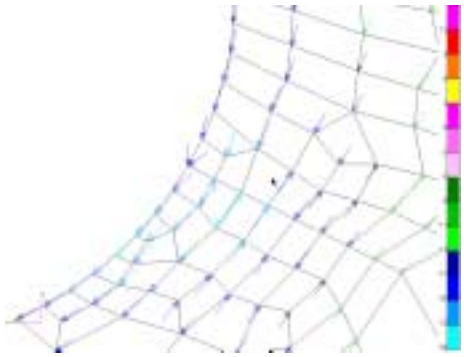


Min Principal a=3

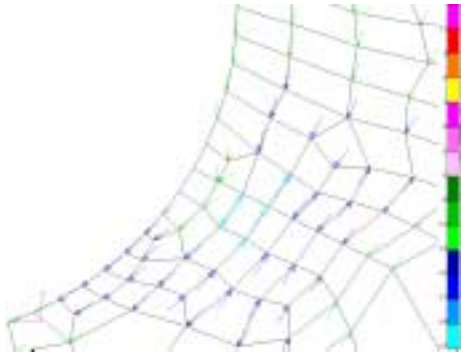
Appendix 3 Nastran Results



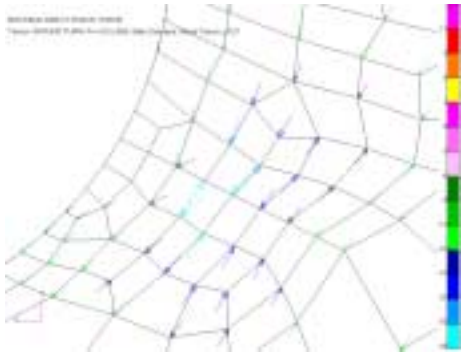
Nastran a=0



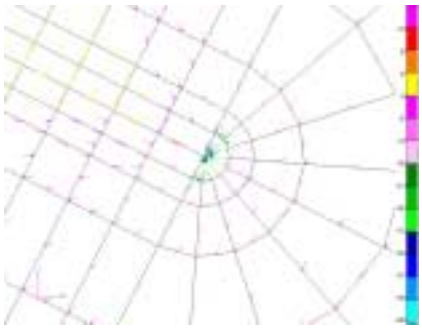
Nastran a=1



Nastran a=2



Nastran a=3



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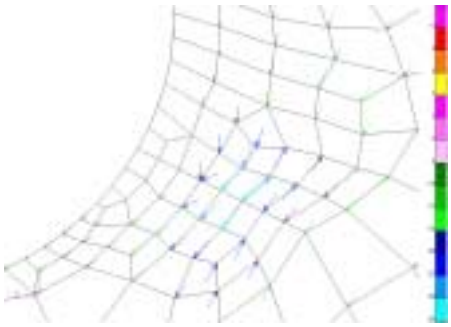

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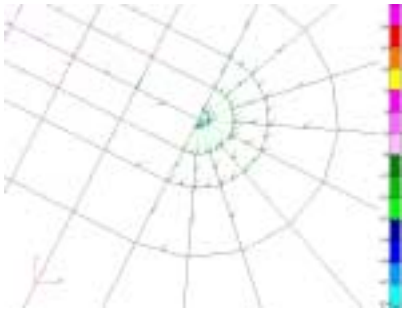

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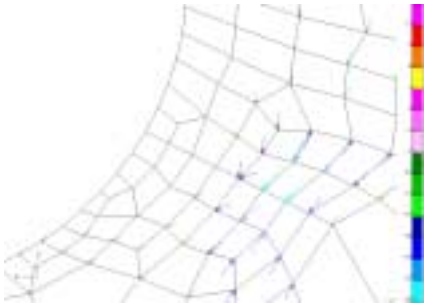

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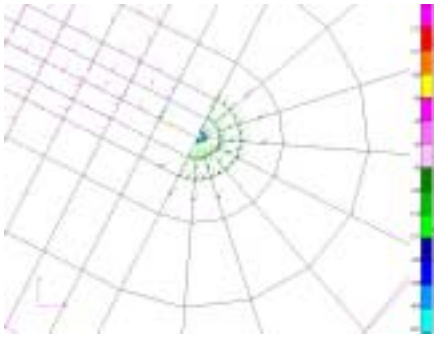

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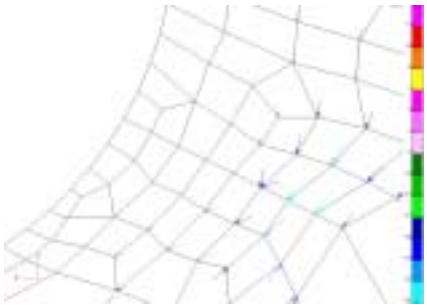

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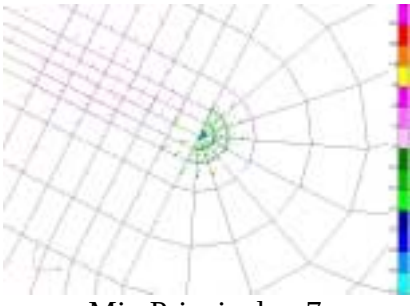

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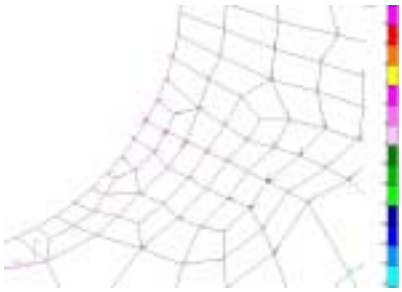
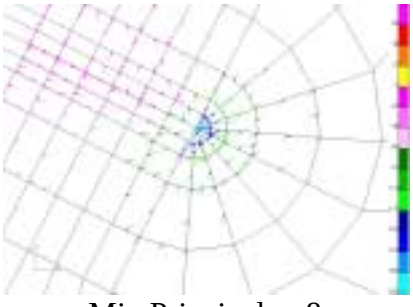


Nastran a=7



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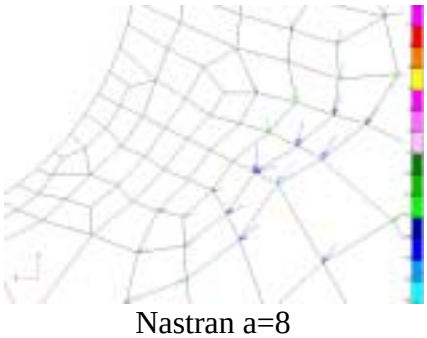
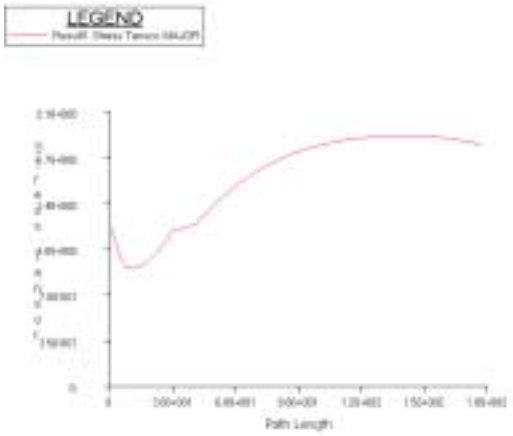
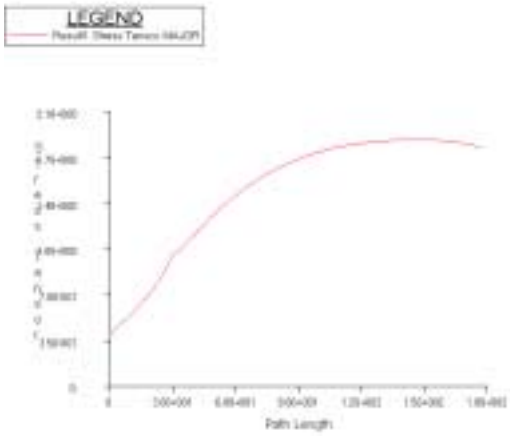
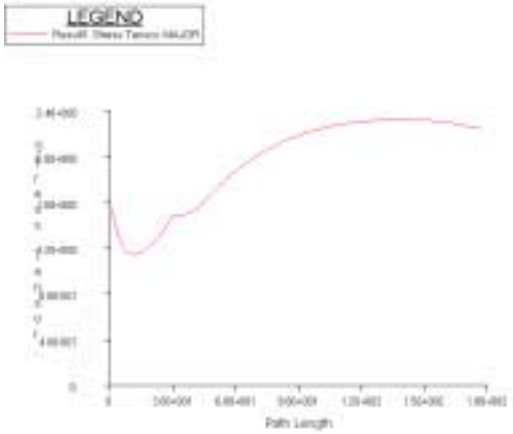
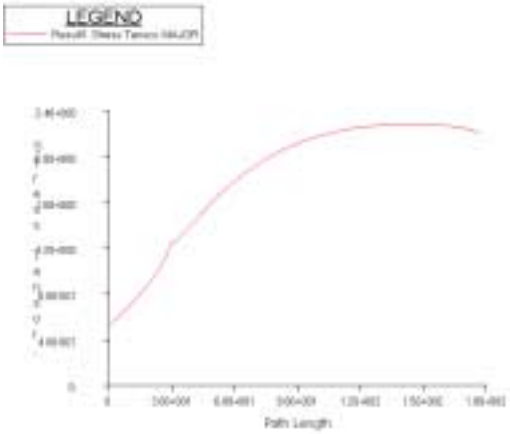
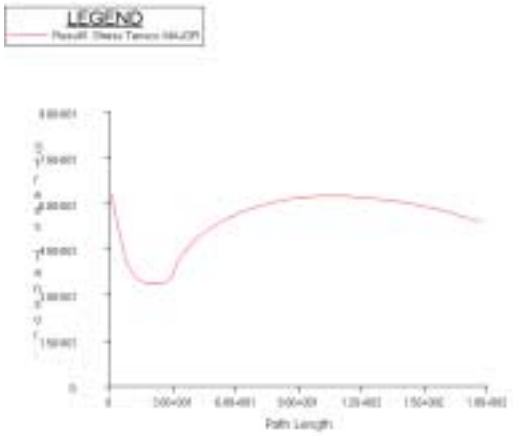
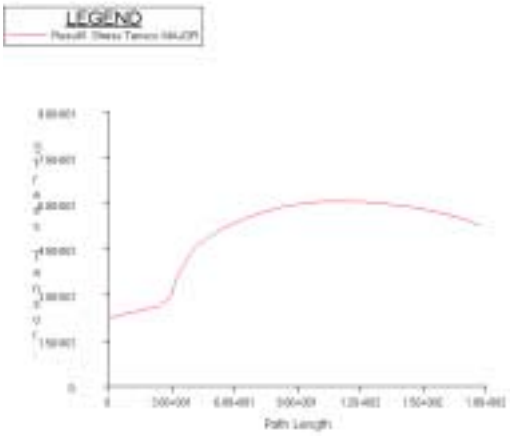
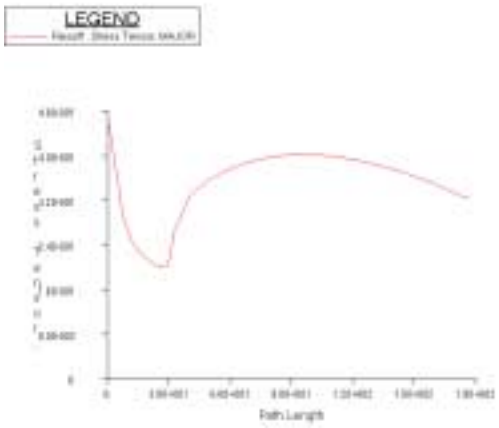


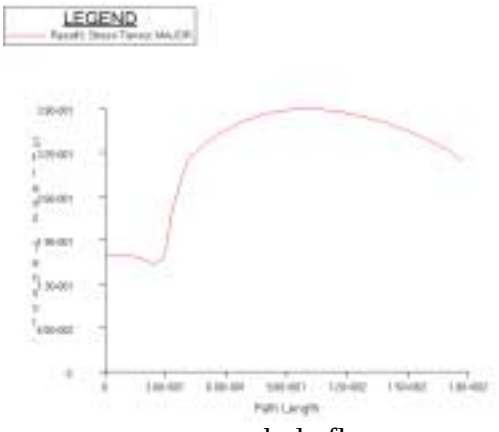
Nastran a=8

Appendix 4 Stress distribution across critical crack plane

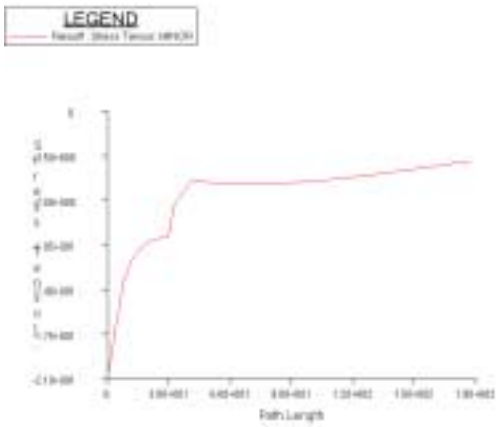
Plate in Figure 2	Plate in Figure 2 with hole removed
 hole climb	 no hole climb 1
 hole cruise 1	 no hole cruise 1
 hole fdapp1	 no hole fdapp1



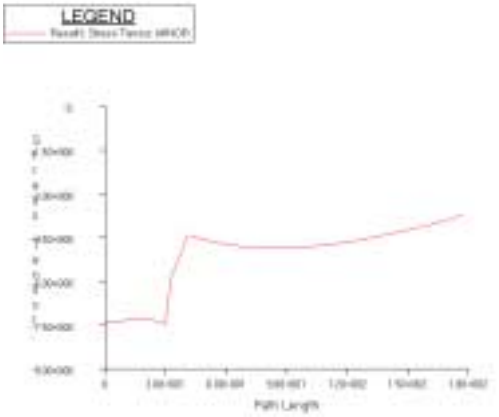
hole flare



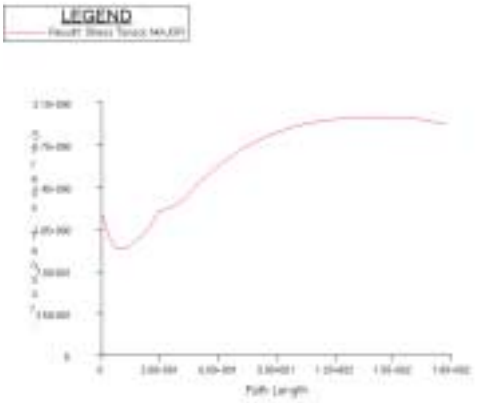
no hole flare



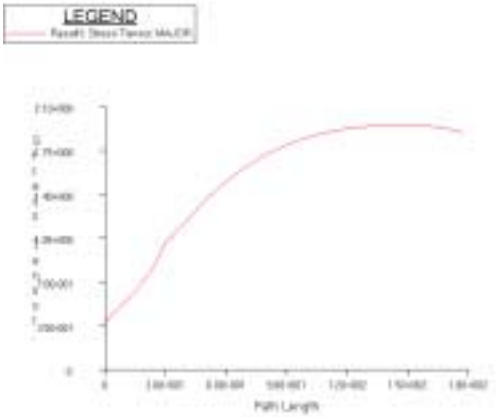
hole GT+15



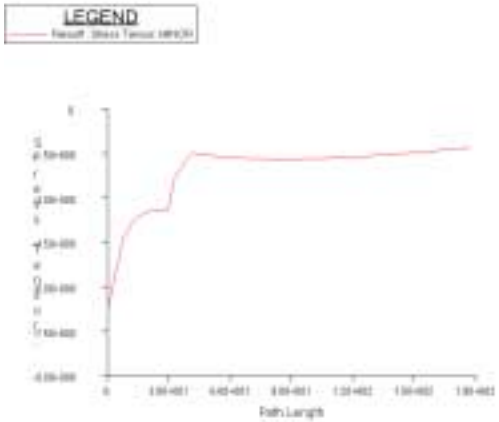
no hole GT+15



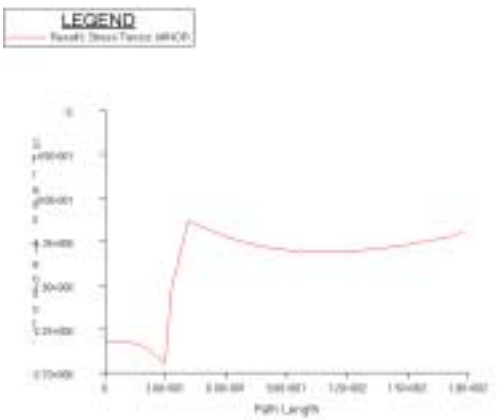
hole init des 1



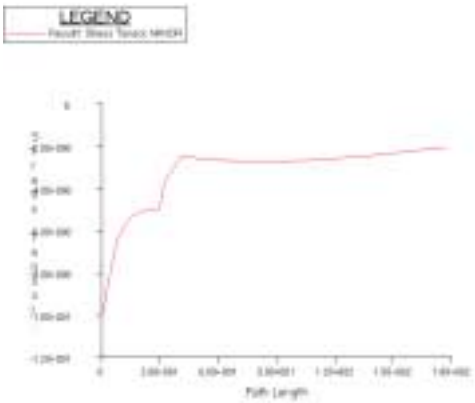
no hole init des 1



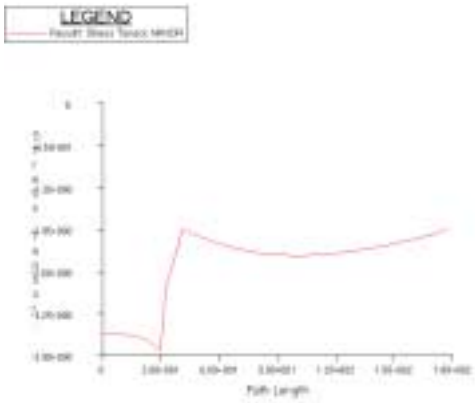
hole rotation



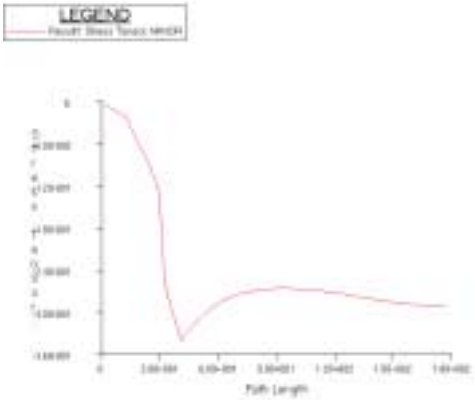
no hole rotation



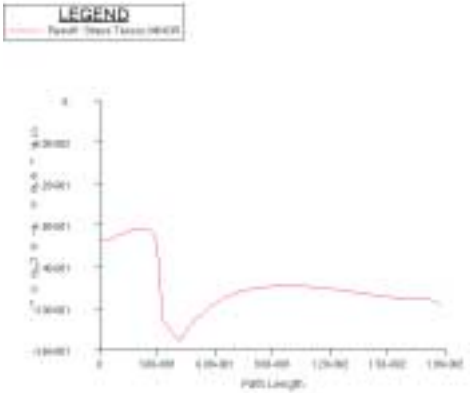
hole taxi 07



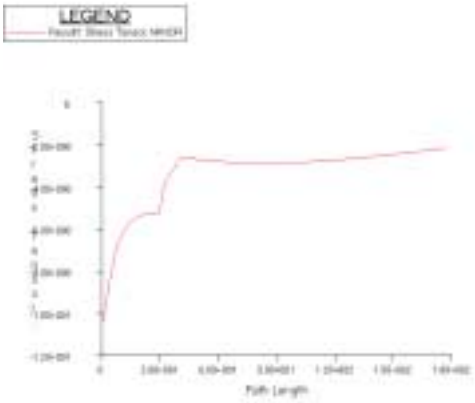
no hole taxi 0.7



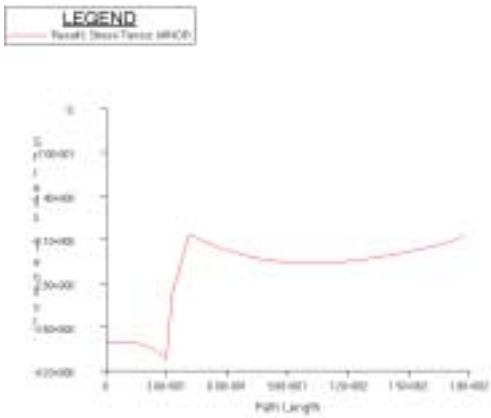
hole TD1



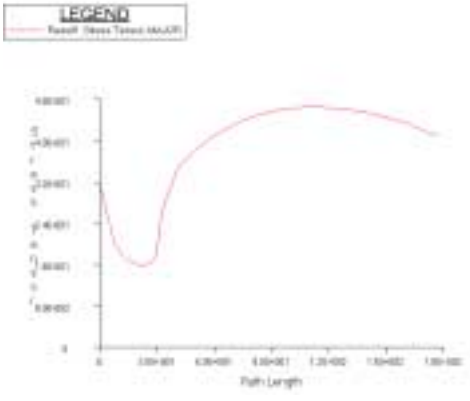
no hole TD1



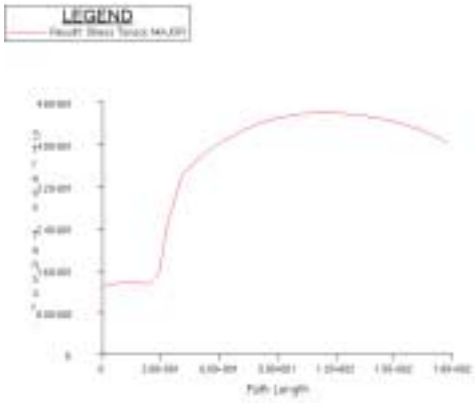
hole unloaded



no hole unloaded



hole yaw1



no hole yaw 1